LANDSAT DATA CONTINUITY MISSION

ENVIRONMENTAL VERIFICATION REQUIREMENTS (LEVR)

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Signature Page

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LDCM PROJECT

DOCUMENT CHANGE RECORD

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REV LEVEL	DESCRIPTION OF CHANGE	APPROVED BY	DATE APPROVED

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SECTION I

GENERAL INFORMATION

1.1 PURPOSE

This document provides requirements and guidelines for the environmental verification program for the Landsat Data Continuity Mission (LDCM) for the observatory (spacecraft), instruments, spacecraft bus, subsystems and components and describes methods for implementing those requirements. It contains a baseline for demonstrating by test or analysis the satisfactory performance of hardware in the expected mission environments, and that minimum workmanship standards have been met. It elaborates on those requirements, gives guideline test levels, provides guidance in the choice of test options, and describes acceptable test and analytical methods for implementing the requirements.

For procurement purposes, both the instruments and the spacecraft "bus" are treated as subsystems. There will be multiple environmental verification plans to address requirements specified within this document at the subsystem and observatory levels.

The General Environmental Verification Specification (GEVS, GSFC-STD-7000) was tailored to create the LDCM project specific verification requirements. .

1.2 APPLICABILITY AND LIMITATIONS

These requirements apply to LDCM hardware and associated software that is to be launched on a Medium Class ELV (e.g. Delta II). These requirements shall apply to all space flight hardware, including interface hardware, that is developed as part of the LDCM project.

These Requirements are written in accordance with the current GSFC practice of using a single protoflight observatory for both qualification testing and space flight (see definition of hardware, 1.8). The protoflight verification program, therefore, is given as the nominal test program.

1.3 THE LDCM VERIFICATION APPROACH

The entire LDCM observatory and its subsystems shall be verified under conditions that simulate the launch operations, flight operations and flight environment as realistically as possible. If the conditions warrant units powered on for launch, those units shall be powered on and their performance monitored during test. However, it is recognized that there may be unavoidable exceptions, or conditions which make it preferable to perform the verification activities at lower levels of assembly. For example, testing at lower levels of assembly may be necessary to produce sufficient environmentally induced stresses to uncover design and workmanship flaws.

Environmental verification of hardware is only a portion of the total assurance effort for LDCM that establishes confidence that the observatory will function correctly and fly a successful mission. The environmental test program provides confidence that the design will perform when subjected to environments more severe than expected during the mission, and provides environmental stress screening to uncover workmanship defects.

The total verification process also includes the development of models representing the hardware, tests to verify the adequacy of the models, analyses, alignments, calibrations, functional/performance tests to verify proper operation, and finally end-to-end tests and simulations to show that the total system will perform as specified.

The level, procedure, and decision criteria for performing any additional tests shall be included in the LDCM system verification plan

1.4 OTHER ASSURANCE REQUIREMENTS

In addition to the verification program, the assurance effort includes parts and materials selection and control, reliability assessment, quality assurance, software assurance, design reviews, and system safety per the LDCM Mission Assurance Requirements (MAR).

1.5 RESPONSIBILITY FOR ADMINISTRATION

The responsibility and authority for decisions in applying the requirements rest with the GSFC LDCM project manager.

The requirements thus derived and deviations from the requirements of this document are subject to review and approval by the GSFC LDCM project office.

1.6 LEVR CONFIGURATION CONTROL AND DISTRIBUTION

This document is controlled and maintained by the GSFC LDCM Project office.

1.7 APPLICABLE DOCUMENTS

The following documents are needed in formulating the environmental test program. The Contractor shall use the current version in effect at the time of procurement for that subsystem.

1.7.1 (deleted)

1.7.2 (deleted)

1.7.3 ELV Payload User Manual - The most recent version of the launch vehicle user manual and requirements are applicable in accordance with the launch vehicle and shall be acquired from the service provider.

1.7.4 (deleted)

1.7.5 Spacecraft Tracking and Data Network Simulation - STDN No. 101.6, Portable Simulation System and Simulations Operation Center Guide for TDRSS & GSTDN, describes the Spacecraft Tracking and Data Network (STDN) and the Tracking and Data Relay Satellite (TDRS)/Ground STDN network simulation programs, and the Simulations Operations Center (SOC). It also discusses end-to-end simulation techniques. STDN No. 408, TDRS and GSTDN Compatibility Test Van Functional Description and Capabilities, describes the equipment and the compatibility test system.

1.7.6 (deleted)

1.7.7 NASA Standards – The following standards provide supporting information:

a. NASA-STD 7002, Payload Test Requirements

b. NASA-STD-7001, Payload Vibroacoustic Test Criteria

c. NASA-STD-7003, Pyroshock Test Criteria

d. NASA-HDBK-7004, Force Limited Vibration Testing

e. NASA-HDBK-7005, Dynamic Environmental Criteria

f. NASA-STD-5001, Structural Design and Test Factors of Safety for Space Flight Hardware

g. NASA-STD-5002, Load Analyses of Spacecraft and Observatories

I.7.8 Military Standards for EMC Testing - Pertinent sections of the following standards are needed to conduct the EMC tests:

a. MIL-STD-461C, Electromagnetic Interference Characteristics Requirements for Equipment.

b. MIL-STD-462, Electromagnetic Interference Characteristics, Measurement of, as amended by Notice I.

c. MIL-STD-463A, Definitions and Systems of Units, Electromagnetic Interference and Electromagnetic Compatibility Technology.

1.7.9 Military Standards for Non-Destructive Evaluation

a. MIL-HDBK-6870, Inspection Program Requirements, Non-Destructive Testing for Aircraft and Missile Materials and Parts.

b. NAS-410, Certification and Qualification of Nondestructive Test Personnel.

c. MSFC-STD-1249, Standard NDE Guidelines and Requirements for Fracture Control Programs.

d. MIL-HDBK-728, Nondestructive Testing.

1.8 DEFINITIONS

The following definitions apply within the context of this specification:

Acceptance Tests: The verification process that demonstrates that hardware is acceptable for flight. It also serves as a quality control screen to detect deficiencies and, normally, to provide the basis for delivery of an item under terms of a contract.

Assembly: See Level of Assembly.

Component: See Level of Assembly.

Configuration: The functional and physical characteristics of the observatory and all its integral parts, assemblies and systems that are capable of fulfilling the fit, form and functional requirements defined by performance specifications and engineering drawings.

Contamination: The presence of materials of molecular or particulate nature which degrade the performance of hardware.

Design Qualification Tests: Tests intended to demonstrate that the test item will function within performance specifications under simulated conditions more severe than those expected from ground handling, launch, and orbital operations. Their purpose is to uncover deficiencies in design and method of manufacture. They are not intended to exceed design safety margins or to introduce unrealistic modes of failure. The design qualification tests may be to either “prototype” or “protoflight” test levels.

Design Specification: Generic designation for a specification that describes functional and physical requirements for an article, usually at the component level or higher levels of assembly. In its initial form, the design specification is a statement of functional requirements with only general coverage of physical and test requirements. The design specification evolves through the project life cycle to reflect progressive refinements in performance, design, configuration, and test requirements. In many projects the end-item specifications serve all the purposes of design specifications for the contract end-items. Design specifications provide the basis for technical and engineering management control.

Electromagnetic Compatibility (EMC): The condition that prevails when various electronic devices are performing their functions according to design in a common electromagnetic environment.

Electromagnetic Interference (EMI): Electromagnetic energy which interrupts, obstructs, or otherwise degrades or limits the effective performance of electrical equipment.

Electromagnetic Susceptibility: Undesired response by a component, subsystem, or system to conducted or radiated electromagnetic emissions.

End-to-End Tests: Tests performed on the integrated ground and flight system, including all elements of the observatory, its control, stimulation, communications, and data processing to demonstrate that the entire system is operating in a manner to fulfill all mission requirements and objectives.

Failure: A departure from specification that is discovered in the functioning or operation of the hardware or software. See nonconformance.

Flight Acceptance: See Acceptance Tests.

Fracture Control Program: A systematic project activity to ensure that a observatory intended for flight has sufficient structural integrity as to present no critical or catastrophic hazard. Also to ensure quality of performance in the structural area for any observatory (spacecraft) project. Central to the program is fracture control analysis, which includes the concepts of fail-safe and safe-life, defined as follows:

a. Fail-safe: Ensures that a structural element, because of structural redundancy, will not cause collapse of the remaining structure or have any detrimental effects on mission performance.

b. Safe-life: Ensures that the largest flaw that could remain undetected after non-destructive examination would not grow to failure during the mission.

Functional Tests: The operation of a unit in accordance with a defined operational procedure to determine whether performance is within the specified requirements.

Hardware: As used in this document, there are two major categories of hardware as follows:

a. Prototype Hardware: Hardware of a new design; it is subject to a design qualification test program; it is not intended for flight.

b. Flight Hardware: Hardware to be used operationally in space. It includes the following subsets:

    (1) Protoflight Hardware: Flight hardware of a new design; it is subject to a qualification test program that combines elements of prototype and flight acceptance verification; that is, the application of design qualification test levels and flight acceptance test durations.

    (2) Follow-On Hardware: Flight hardware built in accordance with a design that has been qualified either as prototype or as protoflight hardware; follow-on hardware is subject to a flight acceptance test program.

    (3) Spare Hardware: Hardware the design of which has been proven in a design qualification test program; it is subject to a flight acceptance test program and is used to replace flight hardware that is no longer acceptable for flight.

Level of Assembly: The environmental test requirements of LEVR generally start at the component or unit level assembly and continue hardware/software build through the system level (referred to in LEVR as the observatory or spacecraft level). The assurance program includes the part level. Verification testing may also include testing at the assembly and subassembly levels of assembly; for test record keeping these levels are combined into a "subassembly" level. The verification program continues through launch, and on-orbit performance. The following levels of assembly are used for describing test and analysis configurations:

Assembly: A functional subdivision of a component consisting of parts or subassemblies that perform functions necessary for the operation of the component as a whole. Examples are a power amplifier and gyroscope.

Bus: An integrated assemblage of modules, subsystems, etc., designed to accommodate the instrument(s) to perform the specified mission in space; i.e. the observatory with no instruments. Another term used to designate this level of assembly is Spacecraft Bus.

Component: A functional subdivision of a subsystem and generally a self-contained combination of items performing a function necessary for the subsystem's operation. Examples are electronic box, transmitter, gyro package, actuator, motor, battery. For the purposes of this document, "component" and "unit" are used interchangeably.

Instrument: An observatory subsystem consisting of sensors and associated hardware for making measurements or observations in space. For the purposes of this document, an instrument is considered a subsystem (of the observatory).

Module: A major subdivision of the observatory that is viewed as a physical and functional entity for the purposes of analysis, manufacturing, testing, and recordkeeping. Examples include spacecraft bus, science instrument, and upper stage vehicle.

Observatory: An integrated assemblage of modules, subsystems, etc., designed to perform a specified mission in space. For the purposes of this document, "payload" and "observatory" are used interchangeably. Other terms used to designate this level of assembly are Laboratory, Satellite and System Segment.

Part: A hardware element that is not normally subject to further subdivision or disassembly without destruction of design use. Examples include resistor, integrated circuit, relay, connector, bolt, and gaskets.

Spacecraft: See Observatory. Other terms used to designate this level of assembly are Laboratory and satellite.

Section: A structurally integrated set of components and integrating hardware that form a subdivision of a subsystem, module, etc. A section forms a testable level of assembly, such as components/units mounted into a structural mounting tray or panel-like assembly, or components that are stacked.

Subassembly: A subdivision of an assembly. Examples are wire harness and loaded printed circuit boards.

Subsystem: A functional subdivision of a observatory consisting of two or more components. Examples are structural, attitude control, electrical power, and communication subsystems. Also included as subsystems of the observatory are the science instruments or experiments.

Unit: A functional subdivision of a subsystem, or instrument, and generally a self-contained combination of items performing a function necessary for the subsystem's operation. Examples are electronic box, transmitter, gyro package, actuator, motor, battery. For the purposes of this document, "component" and "unit" are used interchangeably.

Limit Level: The maximum expected flight level (consistent with the minimum probability levels of Table 2.4-2).

Margin: The amount by which hardware capability exceeds requirements.

Module: See Level of Assembly.

Nonconformance: A condition of any hardware, software, material, or service in which one or more characteristics do not conform to specified requirements.

Offgassing: The emanation of volatile matter of any kind from materials into a manned pressurized volume.

Outgassing: The emanation of volatile materials under vacuum conditions resulting in a mass loss and/or material condensation on nearby surfaces.

Part: See Level of Assembly.

Observatory: See Level of Assembly.

Performance Verification: Determination by test, analysis, or a combination of the two that the observatory element can operate as intended in a particular mission; this includes being satisfied that the design of the observatory or element has been qualified and that the particular item has been accepted as true to the design and ready for flight operations.

Protoflight Testing: See Hardware.

Prototype Testing: See Hardware.

Qualification: See Design Qualification Tests.

Redundancy (of design): The use of more than one independent means of accomplishing a given function.

Section: See Level of Assembly.

Spacecraft: See Level of Assembly.

Subassembly: See Level of Assembly.

Subsystem: See Level of Assembly.

Temperature Cycle: A transition from some initial temperature condition to temperature stabilization at one extreme and then to temperature stabilization at the opposite extreme and returning to the initial temperature condition.

Temperature Stabilization: The condition that exists when the rate of change of temperatures has decreased to the point where the test item may be expected to remain within the specified test tolerance for the necessary duration or where further change is considered acceptable.

Thermal Balance Test: A test conducted to verify the adequacy of the thermal model, the adequacy of the thermal design, and the capability of the thermal control system to maintain thermal conditions within established mission limits.

Thermal-Vacuum Test: A test conducted to demonstrate the capability of the test item to operate satisfactorily in vacuum at temperatures based on those expected for the mission. The test, including the gradient shifts induced by cycling between temperature extremes, can also uncover latent defects in design, parts, and workmanship.

Unit: See Level of Assembly.

Vibroacoustics: An environment induced by high-intensity acoustic noise associated with various segments of the flight profile; it manifests itself throughout the observatory in the form of directly transmitted acoustic excitation and as structure-borne random vibration.

Workmanship Tests: Tests performed during the environmental verification program to verify adequate workmanship in the construction of a test item. It is often necessary to impose stressses beyond those predicted for the mission in order to uncover defects. Thus random vibration tests are conducted specifically to detect bad solder joints, loose or missing fasteners, improperly mounted parts, etc. Cycling between temperature extremes during thermal-vacuum testing and the presence of electromagnetic interference during EMC testing can also reveal the lack of proper construction and adequate workmanship.

1.9 CRITERIA FOR UNSATISFACTORY PERFORMANCE

Deterioration or any change in performance of any test item that does or could in any manner prevent the item from meeting its functional, operational, or design requirements throughout its mission shall be reason to consider the test item as having failed. Other factors concerning failure are considered in the following paragraphs. Further elaboration of project requirements are contained in the LDCM MAR.

1.9.1 Failure Occurrence

When a failure (non-conformance or trend indicating that an out of spec condition will result) occurs, a determination shall be made as to the feasibility and value of continuing the test to its specified conclusion. If corrective action is taken, the test shall be repeated to the extent necessary to demonstrate that the test item's performance is satisfactory.

1.9.2 Failures with Retroactive Effects

If corrective action taken as a result of failure, e.g. redesign of a component, affects the validity of previously completed tests, prior tests shall be repeated to the extent necessary to demonstrate satisfactory performance as deemed required by the MRB.

1.9.3 Failure Reporting

Every failure shall be recorded and reported in accordance with the failure reporting provisions of the GSFC LDCM MAR.

1.9.4 Wear Out

If during a test sequence a test item is operated in excess of design life and wears out or becomes unsuitable for further testing from causes other than deficiencies, a spare may be substituted. If, however, the substitution affects the significance of test results, the test during which the item was replaced and any previously completed tests that are affected shall be repeated to the extent necessary to demonstrate satisfactory performance.

1.10 TEST SAFETY RESPONSIBILITIES

The following paragraphs define the responsibilities shared by the LDCM project and facility management for planning and enforcing industrial safety measures taken during testing for the protection of personnel, the observatory, and the test facility.

1.10.1 Operations Hazard Analysis, Responsibilities For

It shall be the Contractor's responsibility of to ensure that environmental tests and associated operations present no unacceptable hazard to the test item, facilities, or personnel. A test operations hazard analysis (OHA) shall be performed by the Contractor I to consider and evaluate all hazards presented by the interaction of the observatory and the facility for each environmental test. All hazards discovered in the OHA shall be tracked to an agreed-upon resolution. The safety measures to be taken as a result of the OHA, as well as the safety measures between tests, shall be specified as requirements in the verification plan and verification specification. (sec. 2.1.1)

1.10.2 Treatment of Hazards

As hazards are discovered, a considered attempt shall be made to eliminate them. This may be accomplished by redesign, controlling energy sources, revising the test, or by some other method. If the hazard cannot be eliminated, automatic safety controls shall be applied, for example: pressure relief devices, electrical circuit protection devices, or mechanical interlocks. If that is not possible or is too costly, warning devices shall be considered. If none of the foregoing methods are practicable, control procedures shall be developed and applied. In practice, a combination of all four methods may be the best solution to the hazards posed by a complex system. Before any test begins, the Contractor and test facility management shall agree on the hazard control method(s) that are to be used.

1.10.3 Facility Safety

The test facility manager shall verify that the test facility and normal operations present no unacceptable hazard to the test item, test and support equipment, or personnel. He shall ensure that facility personnel abide by all applicable regulations, observe all appropriate industrial safety measures, and follow all requirements for protective equipment. He shall ensure that all facility personnel are trained and qualified for their positions. Training shall include the handling of emergencies by the simulation of emergency conditions. Analyses, tests, and inspections shall be performed to verify that the safety requirements are satisfied. The approach outlined in 1.11.2 shall be used to eliminate or control hazards.

1.10.4 Safety Responsibilities During Tests

The Contractor shall appoint a safety officer to work closely with a safety officer designated by the GSFC LDCM project. The Contractor safety officer shall ensure that the facility meets applicable Occupational Safety & Health Act (OSHA) and other requirements, that appropriate industrial safety measures are observed, and that protective equipment is provided for all personnel involved. The Contractor safety officer shall ensure that the Contractor personnel use the equipment provided and that the test operation does not present a hazard to the facility, space hardware, equipment, or personnel.

1.11 TESTING OF SPARE HARDWARE

A supply of selected spares is often maintained in case of the failure of flight hardware. As a minimum, spares shall undergo a verification program equal to that required for follow-on hardware. Therefore, special consideration shall be given to spares as follows:

a. Extent of Testing - The extent and type of testing shall be determined as part of the flight hardware test program. A spare unit may be used for qualification of the hardware by subjecting it to protoflight testing, and testing the flight hardware to acceptance levels.

b. Spares From Failed Elements - If a flight element is replaced for reasons of failure and is then repaired and redesignated as a spare, appropriate retesting shall be conducted.

c. Caution on the Use of Spares - When the need for a spare arises, immediate analysis and review of the failed hardware shall shall be made. If failure occurs in a hardware item of which there are others of identical design, the fault may be generic and may affect all hardware of that design.

d. "One-Shot" Items - Some items may be degraded or expended during the integration and test period and replaced by spares. The spare that is used shall have met the required quality control standards or auxiliary tests for such items and shall be of qualified design. Examples are pyrotechnic devices, yo-yo despin weights, and elements that absorb impact energy by plastic yielding. When the replacement entails procedures that could jeopardize mission success, the replacement procedure shall be successfully demonstrated with the hardware in the same configuration that it will be in when final replacement is to be accomplished.

1.12 TEST FACILITIES, CALIBRATION

The facilities and fixtures used in conducting tests shall be capable of producing and maintaining the test conditions prescribed with the test specimen installed and operating or not operating, as required. In any major test, facility performance shall be verified prior to the test either by a review of its performance during a test that occurred a short time earlier or by conducting a test with a substitute test item. All equipment used for tests shall be in current calibration and so noted by tags and stickers.

1.13 TEST CONDITION TOLERANCES

In the absence of a rationale for other test condition tolerances, the following shall be used; the values include measurement uncertainties:

Acoustics	Overall Level:	≤ 1 dB	
	l/3 Octave Band Tolerance:	Frequency (Hz)	Tolerance (dB)
		f ≤ 40	+3, -6
		40< F < 3150	± 3
		f ≥ 3150	+3, -6
Antenna Pattern Determination		± 2 dB	
Electromagnetic Compatibility			
	Voltage Magnitude:	± 5% of the peak value	
	Current Magnitude:	± 5% of the peak value	
	RF Amplitudes:	± 2 dB	
	Frequency:	± 2%	
	Distance:	± 5% of specified distance or	
	± 5 cm, whichever is greater		
Humidity		± 5% RH	
Loads	Steady-State (Acceleration):	± 5%	
	Static:	± 5%	
Mass Properties	Weight:		± 0.2%
	Center of Gravity:		± 0.15cm (± 0.06 in.)
	Moments of Inertia:		± 1.5%
Mechanical Shock	Response Spectrum:		+25%, -10%
	Time History:		± 10%

Pressure	Greater than 1.3 X 10^4 Pa (Greater than 100 mm Hg):		± 5%
	1.3 X $l0^4$ to 1.3 X $l0^2$ Pa (l00 mm Hg to 1 mm Hg):		±10%
	1.3 X $l0^2$ to 1.3 X 10^1 Pa (1 mm Hg to 1 micron):		± 25%
	Less than 1.3 X 10^1 Pa (less than 1 micron):		± 80%
Temperature			± 2°C
Vibration	Sinusoidal:	Amplitude	± 10%
		Frequency	±2%
	Random:	RMS level	±10%
		Accel. Spectral Density	± 3 dB

SECTION 2

VERIFICATION PROGRAM

SECTION 2.1

SYSTEM PERFORMANCE VERIFICATION

2.1 SYSTEM PERFORMANCE VERIFICATION

This section applies to the entire LDCM observatory and its subsystems.

2.1.1 Documentation Requirements

The documentation requirements associated with this LEVR as applies to the OLI are called out in the LDCM OLI Contract Data Requirements List.

GEVS Reference	**OLI CDRL**
System Performance Verification Plan	SE-6, System Performance Verification Plan and Matrix
Environmental Verification Plan	IT-6, OLI Environmental Verification Plan and Environmental Test Matrix
System Performance Verification Matrix	SE-6
Environmental Test Matrix	IT-6
Environmental Verification Specification	**This document**
Verification Procedures	Spacecraft, Sensor, and Observatory Test Plans (IT-1, 3, 8)
Verification Reports	SE-7, Verification Reports
System Performance Verification Report	SE-6

The documentation requirements associated with this LEVR as applies to the Spacecraft Bus are called out in the TBD.

SECTION 2.2

ENVIRONMENTAL VERIFICATION

2.2 APPLICABILITY

Sections 2.3 through 2.8 define the LDCM environmental verification program requirements for verifying the observatory, subsystems, and components as follows:

2.3 Electrical Function & Performance
2.4 Structural and Mechanical
2.5 EMC
2.6 Thermal
2.7 Contamination Control
2.8 End-to-End Testing (observatory)

For the purposes of this document, a spacecraft is considered a observatory, an instrument is considered to be a subsystem, a spacecraft bus is considered to be a subsystem when determining the environmental verification requirements.

The basic provisions are written assuming protoflight hardware. They are, in general, also applicable to prototype hardware. Acceptance requirements are also given for the flight acceptance of previously qualified hardware. This applies to follow-on hardware (multiple copies of the same item) developed for the program, or hardware (from another program) qualified by similarity.

2.2.1 Test Sequence and Level of Assembly

The verification activities herein are grouped by discipline; they are not in a recommended sequence of performance. No specific environmental test sequence is required, but the test program shall be arranged in a way to best disclose problems and failures associated with the characteristics of the hardware and the mission objectives.

Table 2.2-1 provides a hierarchy of levels of assembly for the flight hardware, with examples. These level designators are based on those used in the Space Systems Engineering Database developed by The Aerospace Corporation for the Air Force, and agreed to by NASA Headquarters, GSFC, and JPL.. The LEVR environmental test requirements generally start at the “unit” level and end at the “system segment” level. However, screening and life-tests often occur at lower levels, and overall system verification continues beyond the “system segment” level.

2.2.2 Verification Program Tailoring

This document assumes that the observatory is of modular design and can be tested at the unit/component, subsystem/instrument, and observatory system levels of assembly. The Contractor shall develop a verification program that satisfies the intent of the required verification program while taking into consideration the specific characteristics of the mission and the hardware. For example:

- An observatory subsystem, or instrument, may be a functional subdivision of the spacecraft, but it may be distributed throughout the spacecraft rather than being a physical entity. In this case, the environmental tests, and associated functional tests, shall be performed at physical levels of assembly (component, section, module, system or instrument [refer to Appendix A - hardware level of assembly]) that are appropriate for the specific hardware. Performance tests and calibrations may still be performed on the functional subsystem or instrument.

- The physical size of the system may necessitate testing at other levels of assembly. Facility limitations may not allow certain environmental tests to be performed at the system level. In this case, testing shall be performed at the highest practicable level. Also, for very large systems or subsystems/instruments, tests at additional levels of assembly may be added in order to adequately verify the hardware design, workmanship and/or performance.

- For small observatories, the subsystem level environmental tests may be skipped in favor of testing at the component and system/observatory levels. Similarly, for very small instruments the GSFC project may elect to not test all components in favor of testing at the instrument level. These decisions shall be made carefully, especially regarding bypassing lower level testing for instruments, because of the increased risk to the program (schedule, cost, etc.) of finding problems late in the planned schedule.

- In some cases, because of the hardware configuration it may be reasonable to test more than one component at a time. The components may be stacked in their flight configuration, and may therefore be tested as a "section". Part of the decision process shall consider the physical size and mass of the hardware. The test configuration shall allow for adequate dynamic or thermal stress inputs to the hardware to uncover design errors and workmanship flaws.

- Some test requirements stated as subsystem/instrument requirements may be satisfied at a higher level of assembly if approved by the GSFC project. For example, externally induced mechanical shock test requirements may be satisfied at the system level by firing the environment-producing pyro. A simulation of this environment is difficult, especially for large subsystems or instruments.

- Aspects of the design and/or mission may negate certain test conditions to be imposed. For example, if the on-orbit temperature variations are small, less than 5∞C, then consideration shall be given to waiving the thermal-vacuum cycling at the system, or instrument, level of assembly in favor of increasing the hot and cold dwell times.

The same process shall be applied when developing the test plan for an instrument. While testing is required at the instrument component and all-up instrument levels of assembly, additional test levels may be called for because of hardware complexity or physical size.

Table 2.2-1
Flight System Hardware
Levels of Assembly

LEVEL OF ASSEMBLY	EXAMPLES
Space System	NASA Spacecraft
Project or Program	TDRS TIROS GOES
Operating System	Operating Space System
Integrated Systems	Integrated Flight System (Spacecraft + Upperstage + Launch Vehicle)
System Segment (Satellite, Payload, Spacecraft, Laboratory, Observatory, Space Vehicle, etc.)	(Spacecraft Bus + Science Payload) Launch Vehicle IUS
Module	Spacecraft Bus Science Payload Payload Fairing
Subsystem	Instrument/Experiment, Structure, Attitude Control, C & DH, Thermal Control, Electrical Power, TT & C, Propulsion
Section (group of units/components not a subsystem)	Electronic Tray or Pallette, Stacked Units/Components Electronic Boxes Mounted on Panel, Solar Array Sections
Unit (Component)	Electronic Box, Gyro Package, Motor, Actuator, Battery, Receiver, Transmitter, Antenna, Solar Panel, Valve Regulator
Subassembly (combines assembly and subassembly)	Assembly (Power Amplifier, Gyroscope) Subassembly (Wire Harness, Loaded Printed Circuit Card)
Part	Resistor, Capacitor, IC, Switch, Connector Bolt, Screw, Gasket, Bracket, Valve Stem

.

2.2.3 Qualification of Hardware by Similarity

There are cases in which hardware qualified for one flight program is to be built and used on another program. Hardware that has been previously qualified may be considered qualified for use on a new program by showing that the hardware is sufficiently similar to the original hardware and that the previous qualification program has adequately enveloped the new mission environments. The details for performing this comparison shall be defined by the project but as a minimum the following areas shall be reviewed and documented:

(1) Design and test requirements shall be shown to envelope the original requirements. This shall include a review of the test configuration and of all waivers and deviations that may have occurred during testing of the original hardware.

(2) Manufacturing information shall be reviewed to determine if changes have been made that would invalidate the previous hardware qualification. This review shall cover parts, materials, packaging techniques as well as changes to the assembly process or procedures.

(3) Test experience with the previous flight build shall be reviewed to verify that no significant modifications were made to the hardware during testing to successfully complete the test program. Any significant change shall be identified and shown to be implemented on the current flight hardware.

If the review of the above criteria by the GSFC LDCM project shows that the hardware is of sufficiently similar design as the first build and that the previous test requirements envelope any new environmental requirements, then the hardware can be treated as qualified and need only to be subjected to acceptance level test requirements. The Contractor shall include the review shall documents as part of the verification package.

2.2.4 Test Factors/Durations

Test factors for prototype, protoflight, and acceptance are given in Table 2.2-2.

2.2.5 Structural Analysis/Design Factors of Safety

Structural and mechanical verification testing shall be supported by structural analysis to provide confidence that the hardware will not experience failure or detrimental permanent deformation under test or launch conditions. The factors of safety that shall be applied to limit loads in order to calculate structural margins are shown in Table 2.2-3. These factors of safety have been selected to be consistent with the test factors shown in Table 2.2-2. The yield factor of safety ensures that a prototype or protoflight test can be conducted with low risk of the hardware experiencing detrimental yielding. The ultimate factor of safety provides adequate separation between yield and ultimate failure modes and ensures that the hardware will not experience an ultimate failure under expected loading conditions.

Table 2.2-2
Test Factors/Durations

Test	Prototype Qualification	Protoflight Qualification	Acceptance
Structural Loads[1] Level Duration Centrifuge/Static Load Sine Burst	 1.25 x Limit Load 1 minute 5 cycles @ full level per axis	 1.25 x Limit Load 30 seconds 5 cycles @ full level per axis	 1.0 x Limit Load 30 seconds 5 cycles @ full level per axis
Acoustics Level[2] Duration	Limit Level + 3dB 2 minutes	Limit Level + 3dB 1 minute	Limit Level 1 minute
Random Vibration Level[2] Duration	Limit Level + 3dB 2 minutes/axis	Limit Level + 3dB 1 minute/axis	Limit Level 1 minute/axis
Sine Vibration[3] Level Sweep Rate	1.25 x Limit Level 2 oct/min	1.25 x Limit Level 4 oct/min	Limit Level 4 oct/min
Mechanical Shock Actual Device Simulated	 2 actuations 1.4 x Limit Level 2 x Each Axis	 2 actuations 1.4 x Limit Level 1 x Each Axis	 1 actuations Limit Level 1 x Each Axis
Thermal-Vacuum	Max./min. predict. ± 10°C	Max./min. predict. ± 10°C	Max./min. predict. ± 5°C
Thermal Cycling[4]	Max./min. predict. ± 25°C	Max./min. predict. ± 25°C	Max./min. predict. ± 20°C
EMC & Magnetics	As Specified for Mission	Same	Same

1 - If qualified by analysis only, positive margins shall be shown for factors of safety of 2.0 on yield and 2.6 on ultimate. Beryllium and composite materials cannot be qualified by analysis alone.

Note: Test levels for beryllium structure are 1.4 x Limit Level for both qualification and acceptance testing. Also composite structure, including metal matrix, requires acceptance testing to 1.25 x Limit Level.

2 - As a minimum, the test level shall be equal to or greater than the workmanship level.

3 - The sweep direction shall be evaluated and chosen to minimize the risk of damage to the hardware. If a sine sweep is used to satisfy the loads or other requirements, rather than to simulate an oscillatory mission environment, a faster sweep rate may be considered, e.g., 6-8 oct/min to reduce the potential for over stress.

4 - It is recommended that the number of thermal cycles and dwell times be increased by 50% for thermal cycle (ambient pressure) testing.

Table 2.2-3
Flight Hardware Design/Analysis Factors of Safety Applied to Limit Loads [1,2]

Type	Static	Sine	Random/Acoustic[4]
Metallic Yield	1.25[3]	1.25	1.6
Metallic Ultimate	1.4[3]	1.4	1.8
Stability Ultimate	1.4	1.4	1.8
Beryllium Yield	1.4	1.4	1.8
Beryllium Ultimate	1.6	1.6	2.0
Composite Ultimate	1.5	1.5	1.9
Bonded Inserts/Joints Ultimate	1.5	1.5	1.9

1 – Factors of safety for pressurized systems to be compliant with AFSPCMAN 91-710 (Range safety).

2 – Factors of safety for glass and structural glass bonds specified in NASA-STD-5001

3 – If qualified by analysis only, positive margin shall be shown for factors of safety of 2.0 on yield and 2.6 on ultimate. See section 2.4.1.1.1

4 – Factors shown shall be applied to statistically derived peak response based on RMS level. As a minimum, the peak response shall be calculated as a 3-sigma value.

SECTION 2.3

ELECTRICAL FUNCTION & PERFORMANCE

2.3 ELECTRICAL FUNCTION TEST REQUIREMENTS

The following paragraphs describe the required electrical functional and performance tests that verify the observatory's operation before, during, and after environmental testing. These tests along with all other calibrations, functional/performance tests, measurements/ demonstrations, alignments (and alignment verifications), end-to-end tests, simulations, etc., that are part of the overall verification program shall be described in the System Performance Verification Plan.

2.3.1 Electrical Interface Tests

Before the integration of an assembly, component, or subsystem into the next higher hardware assembly, electrical interface tests shall be performed to verify that all interface signals are within acceptable limits of applicable performance specifications.

Prior to mating with other hardware, electrical harnessing shall be tested to verify proper characteristics; such as, routing of electrical signals, impedance, isolation, and overall workmanship.

The following parameters shall be verified as a minimum:

a. Accuracy (signals on correct pins and nowhere else),
b. Inputs and outputs (unloaded and loaded),
c. Specified range (high/low extremes as well as nominal),
d. Range impacts (how range extremes of one signal affect related signals).

2.3.1.1 Aliveness Tests

An aliveness test shall be performed as necessary to verify that the subsystem and/or observatory and its major components are functioning.

2.3.2 Comprehensive Performance Tests

A comprehensive performance test (CPT) shall be conducted on each hardware element after each stage of assembly: component, subsystem and observatory. When environmental testing is performed at a given level of assembly, additional comprehensive performance tests shall be conducted during the hot and cold extremes of the temperature or thermal-vacuum test for both maximum and minimum input voltage, and at the conclusion of the environmental test sequence, as well as at other times prescribed in the verification plan, specification, and procedures.

The comprehensive performance test shall be a detailed demonstration that the hardware and software meet their performance requirements within allowable tolerances. The test shall demonstrate operation of all redundant circuitry and satisfactory performance in all operational modes within practical limits of cost, schedule, and environmental simulation capabilities. The initial CPT shall serve as a baseline against which the results of all later CPTs can be readily compared.

At the observatory level, the comprehensive performance test shall demonstrate that, with the application of known stimuli, the observatory will produce the expected responses. At lower levels of assembly, the test shall demonstrate that, when provided with appropriate inputs, internal performance is satisfactory and outputs are within acceptable limits.

2.3.3 Limited Performance Tests

Limited performance tests (LPT) shall be performed before, during, and after environmental tests, as appropriate, in order to demonstrate that functional capability has not been degraded by the tests. The limited tests are also used in cases where comprehensive performance testing is not warranted or not practicable. LPTs shall demonstrate that the performance of selected hardware and software functions is within acceptable limits. Specific times when LPTs will be performed shall be prescribed in the verification specification.

2.3.4 Performance Operating Time and Failure-Free Performance Testing

One-thousand (1000) hours of operating/power-on time shall be accumulated on all flight electronic hardware, and spares prior to launch.

In addition, at the conclusion of the performance verification program, the observatory shall have demonstrated failure-free performance testing for at least the last 350 hours of operation. The demonstration may be conducted at the subsystem level of assembly when observatory integration is accomplished at the launch site and the 350-hour demonstration cannot practicably be accomplished on the integrated observatory. Failure-free operation during the thermal-vacuum test exposure is included as part of the demonstration with 100 hours of the trouble-free operation being logged at the hot-dwell temperatures and 100 hours being logged at the cold-dwell temperature. The 350- hour demonstration shall include at least 200 hours in vacuum. Major hardware changes during or after the verification program shall invalidate previous demonstration.

2.3.5 Limited-Life Electrical Elements

A life test program shall be considered for electrical elements that have limited lifetimes. The verification plan shall address the life test program, identifying the electrical elements that require such testing, describing the test hardware that will be used, and the test methods that will be employed.

2.3.6 Long Duration and Failure Free System Level Test of Flight Software

Ground test of the fully integrated FSW system shall include demonstration of error-free operations-like scenarios over an extended time period. The minimum duration uninterrupted FSW system-level test (on the highest fidelity FSW testbed) is 72 hours for Class A and B, 48 hours for Class C, and 36 hours for Class D missions, respectively.

SECTION 2.4

STRUCTURAL AND MECHANICAL

2.4 STRUCTURAL AND MECHANICAL VERIFICATION REQUIREMENTS

A series of tests and analyses shall be conducted to demonstrate that the flight hardware is qualified for the expected mission environments and that the design of the hardware complies with the specified verification requirements such as factors of safety, interface compatibility, structural reliability, workmanship, and associated elements of system safety.

Table 2.4-1 specifies the structural and mechanical verification activities. When the tests and analyses are planned, consideration shall be given to the expected environments of structural loads, vibroacoustics, sine vibration, mechanical shock, and pressure profiles induced during all phases of the mission; for example, during launch, insertion into final orbit, preparation for orbital operations. Verification shall also be accomplished to ensure that the transportation and handling environments are enveloped by the expected mission environments. Mass properties and proper mechanical functioning shall also be verified.

Of equal importance with qualifying the hardware for expected mission environments are the testing for workmanship and structural reliability, which are intended to provide a high probability of proper operation during the mission. In some cases, the expected mission environment is rather benign and produces test levels insufficient to expose workmanship defects. The verification test shall envelope the expected mission levels, with appropriate margins added for qualification, and impose sufficient stress to detect workmanship faults. Flight load and dynamic environment levels are probabilistic quantities. Selection of probability levels for flight limit level loads/environments to be used for observatory design and testing is the responsibility of the Contractor, but in no event shall the probability levels be less than the minimum levels in Table 2.4-2. Specific structural reliability requirements regarding fracture control for ELV payloads, beryllium structure, composite structure, bonded structural joints, and glass structural elements are given in 2.4.1.4.

The program outlined in Table 2.4-1 assumes that the observatory is sufficiently modularized to permit realistic environmental exposures at the subsystem level. When that is not possible, with the GSFC LDCM project office's approval, compliance with the subsystem requirements shall be accomplished at a higher or lower level of assembly. For example, structural load tests of some components may be necessary if they cannot be properly applied during testing at higher levels of assembly.

Ground handling, transportation and test fixtures shall be analyzed and tested for proper strength as required by safety, and shall be verified for stability for applicable configurations as appropriate.

2.4.1 Structural Loads Qualification

Qualification of the observatory for the structural loads environment requires a combination of test and analysis. A test-verified finite element model of the observatory shall be developed and a coupled loads analysis of the observatory/launch vehicle performed.

The analytical results define the limit loads for the observatory (subsystems and components) and show compatibility with the launch vehicle for all critical phases of the mission.

TABLE 2.4-1
Structural and Mechanical Verification Test Requirements

Requirement	Observatory/ Spacecraft	Subsystem/ Instrument	Unit (Component) Including Instrument Units (Components)
Structural Loads			
Modal Survey	*	T^2	*
Design Qualification	*	$A,T/A^1$	*
Structural Reliability Primary & Secondary Structure	*	$(A,T)^1$	*
Vibroacoustics			
Acoustics	T	T^2	T^2
Random Vibration	T^2	T^2	T
Sine Vibration	T^3,T^4	T^3,T^5	T^3,T^6
Mechanical Shock	T	T^7	T^7
Mechanical Function	A,T	A,T	-
Pressure Profile	-	A,T^2	A
Mass Properties	A/T	A,T^2	*

* = May be performed at observatory or component level of assembly if appropriate.

A = Analysis required.

T = Test required.

A/T = Analysis and/or test.

$A,T/A^1$ = Analysis and Test or analysis only if no-test factors of safety given in 2.4.1.1.1 are used.

$(A,T)^1$ = Combination of fracture analysis and proof tests on selected elements, with special attention given to beryllium, composites, and bonded joints.

T^2 = Test shall be performed unless assessment justifies deletion.

T^3 = Test performed to simulate any sustained periodic mission environment, or to satisfy other requirement (loads, low frequency transient vibration).

T^4 = Test shall be performed for ELV observatory, if practicable, to simulate transient and any sustained periodic vibration mission environment.

T^5 = Test shall be performed for ELV observatory instruments and for ELV observatory subsystems if not performed at observatory level of assembly due to test facility limitations; to simulate sine transient and any sustained periodic vibration mission environment.

T^6 = Test shall be performed for ELV observatory, instruments, and components to simulate sine transient and any sustained periodic vibration mission environment.

T^7 = Test required for self-induced shocks, but may be performed at observatory level of assembly for externally induced shocks.

TABLE 2.4-2
Minimum Probability-Level Requirements
for Flight Limit (maximum expected) Level

Requirement	Minimum Probability Level	
		ELV Observatory
Structural Loads		97.72/50 (1),(2)
Vibroacoustics Acoustics Random Vibration		95/50 (3)
Sine Vibration		97.72/50 (1)
Mechanical Shock		95/50

Notes:

(1) When parametric statistical methods are used to determine the limit level, the data shall be tested to show a satisfactory fit to the assumed underlying distribution.

(2) 97.72% probability of not exceeding level, estimated with 50% confidence. Equal to the mean plus two-sigma level for normal distributions.

(3) Equal to, or greater than, the ninety-fifth percentile value, estimated with 50% confidence.

A modal survey shall be performed for each observatory (at the subsystem/instrument or other appropriate level of assembly) to verify that the analytical model adequately represents the dynamic behavior of the hardware. The test-verified model shall then be used to predict the maximum expected load for each critical loading condition, including handling and transportation, vibroacoustic effects during lift-off, insertion into final orbit, orbital operations, etc., as appropriate for the particular mission. If the observatory configuration is different for various phases of the mission, the structural loads qualification program, including the modal survey, shall consider the different configurations. The maximum loads resulting from the analysis shall define the limit loads.

The launch loads environment is made up of a combination of steady-state, low-frequency transient, and higher-frequency vibroacoustic loads. To determine the combined loads for any phase of the launch the root-sum-square (RSS) of the low- and high-frequency dynamic components are superimposed upon the steady-state component if appropriate.

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$$N_i = S_i \_ [(L_i)^2 + (R_i)^2]^{1/2}$$

where N_i, S_i, L_i, and R_i are the combined load factor, steady-state load factor, low-frequency dynamic load factor, and high-frequency random vibration load factor, respectively, for the i'th axis. In some cases, the steady-state and low-frequency dynamic load factors are combined into a low-frequency transient load factor A_i. In this case, the steady-state value shall be separated out before the RSS operation.

When determining the limit loads for ELV launches, consideration shall be given to the timing of the loading events; the maximum steady state and dynamic events occur at different times in the launch and may provide too conservative an estimate if combined. Also, the frequency band of the vibroacoustic energy to be combined shall be evaluated on a case-by-case basis. Flight events which shall be considered for inclusion in the coupled loads analysis for various ELV's are listed in Table 2.4-3. If the verification cycle analysis or observatory test-verified model is not available, the latest analytical data shall be used in conjunction with an uncertainty factor to be approved by the GSFC LDCM project office.

Each subsystem/instrument shall then be qualified by loads testing to 1.25 times the limit loads defined above. The loads test shall be accompanied by stress analysis showing positive margins of safety at 1.4 times the limit load for all ultimate failure modes such as fracture or buckling. In some cases, qualification by analysis may be allowed.

Special design and test factors of safety are required for beryllium structure (see 2.4.1.3.1).

2.4.1.1 Coupled load analysis - A coupled load analysis, combining the launch vehicle and observatory, shall be performed to support the verification of positive stress margins and sufficient clearances during the launch.

2.4.1.1.1 Analysis - Strength Verification - A finite element model shall be developed (and verified by test) that analytically simulates the observatory's mass and stiffness characteristics, for the purpose of performing a coupled loads analysis. The model shall be of sufficient detail to make possible an analysis that defines the observatory's modal frequencies and displacements below a specified frequency that is dependent on the fidelity of the launch vehicle finite element model. For ELV all significant modes below 70 Hz are sufficient unless higher-frequency modes are required by the launch vehicle manufacturer.

The model is then coupled with the model of the ELV and any upper-stage propulsion system. The combined coupled model is used to conduct a coupled loads analysis that evaluates all potentially critical loading conditions. Forcing functions used in the coupled loads analysis shall be defined at the flight limit level consistent with the minimum probability levels of Table 2.4-2. The results of the coupled loads analysis shall be reviewed to determine the worst-case loads. These constitute the set of limit loads that are used to evaluate member loads and stresses.

For ELV observatories, the coupled loads analysis shall consider all flight events required by the ELV provider. None of the flight events shall be deleted from the coupled loads analysis unless it is shown by base drive analysis of the cantilevered observatory and adapter that there are no significant observatory vibration modes in frequency bands of significant launch vehicle forcing functions and coupled-mode responses. For example, it shall be confirmed that there are no observatory structural

components or subsystems (upper platforms, antenna supports, scientific instruments, etc.) which can experience high dynamic responses during flight events such as lift-off or sustained, pogo-like oscillations before deleting these events. For the evaluation of flight events to include in the coupled loads analysis, an appropriate tolerance shall be applied to all potentially significant observatory modal frequencies unless verified by modal survey testing
Normally, the design and verification of observatories shall not be burdened by transportation and handling environments that exceed stresses expected during launch, orbit. Rather, shipping containers shall be designed to prevent the imposition of such stresses. To verify this, a documented analysis shall be prepared on shipping and handling equipment to define the loads transmitted to flight hardware. When transportation and handling loads are not enveloped by the maximum expected flight loads, the transportation and handling loads shall be included in the set of limit loads.

For those hardware items that will later be subjected to a strength qualification test, a stress analysis shall be performed to provide confidence that the risk of failing the strength test is small and to demonstrate compliance with the launch vehicle (ELV) interface verification and safety requirements. The analysis shall show positive margins at stresses corresponding to a loading of 1.4 times the limit load for all ultimate failure modes such as fracture or buckling. In addition, the analysis shall show that for a loading equal to the limit load, the maximum allowable loads at the ELV flight adapter are not exceeded, that no detrimental permanent deformations will occur, and that no excessive deformations occur that might constitute a hazard to the launch vehicle. See 2.4.1.4 for special requirements for beryllium structure.

For observatories, or observatory elements, whose strength is qualified by analysis, the objective of the stress analysis is to demonstrate with a high degree of confidence that there is essentially no chance of failure during flight. For all elements that are to be qualified by analysis, positive strength margins on yield shall be shown to exist at stresses equal to 2.0 times those induced by the limit loads, and positive margins on ultimate shall be shown to exist at stresses equal to 2.6 times those induced by the limit loads. For exceptions, see 2.4.1.3. When qualification by analysis is used, the upper frequency of the modal survey shall be increased by a value as agreed to by the GSFC LDCM project office. In addition, at stresses equal to the limit load, the analysis shall show that the maximum allowable loads at the ELV flight adapter are not exceeded, that no detrimental permanent deformations will occur, and that no excessive deformations occur that might constitute a hazard to the launch vehicle.

2.4.1.1.2 Analysis - Clearance Verification - Analysis shall be conducted for ELV observatories to verify adequate dynamic clearances between the observatory and launch vehicle and between members within the observatory for all significant ground test and flight conditions.

a. During Powered Flight - The coupled loads analysis shall be used to verify adequate clearances during flight within the ELV payload fairing. One part of he coupled loads analysis output transformation matrices shall contain displacement data that will allow calculation of loss of clearance between critical extremities of the observatory and adjacent surfaces of the ELV. For ELV observatories, the analysis shall consider clearances between the observatory and ELV payload fairing (and its acoustic blankets if used, including blanket expansion due to venting) and between the observatory and ELV attach fitting, as applicable. For the clearance calculations the following factors shall be considered:

1. Worst-case observatory and vehicle manufacturing and assembly tolerances as derived from as-built engineering drawings.

2. Worst-case observatory/vehicle integration "stacking" tolerances related to interface mating surface parallelism, perpendicularity and concentricity, plus bolt positional tolerances, ELV payload fairing ovality, etc.

3. Quasi-static and dynamic flight loads, including coupled steady-state and transient sinusoidal vibration, vibroacoustics and venting loads, as applicable. Typically, either liftoff or the transonic buffet and maximum airloads cause the greatest relative deflections between the vehicle and observatory.

b. <u>During ELV Payload Fairing Separation</u> - A fairing separation analysis based on ground separation test of the fairing, shall be used to verify adequate clearances between the separating fairing sections and observatory extremities. Effects of fairing section shell-mode oscillations, fairing rocking, vehicle residual rates, transient coupled-mode oscillations, thrust accelerations, and vehicle control-jet firings shall be considered, as applicable.

c. <u>During Payload Separation</u> - A payload separation analysis shall be used to verify adequate clearances between the observatory and ELV during separation. The analysis shall include effects of factors such as vehicle residual rates, forces and impulses imparted by the separation system (including lateral impulses due to separation clampbands) and vehicle retro-rocket plumes impinging on the observatory, as applicable. The same analysis shall be utilized to verify acceptable payload separation velocity and tip-off rates if required

Analysis shall also be performed to verify adequate critical dynamic clearances between members within the observatory during ground vibration and acoustic testing, and flight. Additionally, a deployment analysis shall be used to verify adequate clearances during observatory appendage deployment. Refer to 2.4.5.2 regarding mechanical function clearances.

For all of the above clearance analyses and conditions, adequate clearances shall be verified assuming worst-case static clearances due to manufacturing, assembly and vehicle integration tolerances (unless measured on the launch stand), and quasi-static and dynamic deflections due to 1.4 times the applicable flight limit loads or flight-level ground test levels. Depending on the available static clearance, the clearance analysis requirements may be satisfied in many cases by simple worst-case estimates and/or similarity.

2.4.1.2 <u>Modal Survey</u> - A modal survey test will be required for observatories and subsystems, including instruments, that do not meet requirements on minimum fundamental frequency. The minimum fundamental frequency requirement is dependent on the launch vehicle and is discussed below for ELV launch vehicles. In order to determine if the hardware meets the frequency requirement, an appropriate test, or tests, shall be performed to identify the fundamental frequency. A low level sine survey is generally an appropriate method for determining the fundamental frequency.

For an ELV, the frequency below which a modal test is required is dependent on the specific launch vehicle. The determination will be made on a case-by-case basis and specified in the design and test requirements. Modal tests shall be performed at the subsystem/instrument level of assembly, but may be required at other levels of assembly such as the observatory or component level depending on LDCM project office determination.

In general, the support of the hardware during the test shall duplicate the boundary conditions expected during launch. When that is not feasible, other boundary conditions are employed and the frequency limits of the test are adjusted by a value as agreed to by the GSFC LDCM project office. The effects of interface flexibilities shall be considered when other than normal boundary conditions are used.

The results of the modal survey are required to identify any inaccuracies in the mathematical model used in the observatory analysis program so that modifications can be made if needed. Such an experimental verification is required because a degree of uncertainty exists in unverified models owing to assumptions inherent in the modeling process. These lead to uncertainties in the results of the flight dynamic loads analysis, thereby reducing confidence in the accuracy of the set of limit loads derived therefrom.

All significant modes (those with greater than 5% effective modal mass) up to the required frequency shall be determined both in terms of frequency and mode shape. Cross-orthogonality checks of the test and analytical mode shapes, with respect to the analytical mass matrix, shall be performed with the goal of obtaining at least 0.9 on the diagonal and no greater than 0.1 off-diagonal. Any test method that is capable of meeting the test objectives with the necessary accuracy may be used to perform the modal survey. The input forcing function may be transient, fixed frequency, swept sinewave, or random in nature.

2.4.1.3 Design Strength Qualification - The preferred method of verifying adequate strength is to apply a set of loads that will generate forces in the hardware that are equal to 1.25 times limit loads. The strength qualification test shall be shown to produce forces equal to 1.25 times limit at structural interfaces as well as in structural elements which have been shown to have the lowest margins for all identified failure modes of the hardware. As many test conditions as necessary shall be applied to achieve the appropriate loads for qualification. Structural qualification testing shall be performed at the lowest level of assembly as possible to reduce overtest and to limit the risk of damage to other components/subsystems shall structural failure occur. After structural testing, the hardware shall be capable of meeting its performance criteria (see 2.4.1.3.1 for special requirements for beryllium structure). No detrimental permanent deformation shall be allowed to occur as a result of applying the loads, and all applicable alignment requirements shall be met following the test.

The strength qualification test shall be accompanied by a stress analysis that demonstrates a positive margin on ultimate at loads equal to 1.4 times the limit load for all ultimate failure modes such as fracture or buckling. See 2.4.1.3.1 for special requirements for beryllium structure.

In addition, the analysis shall show that at stresses equal to the limit load, the maximum allowable loads at the launch vehicle interface points are not exceeded and that no excessive deformations occur that might constitute a hazard to the mission. This analysis shall be performed prior to the start of the strength qualification tests to provide minimal risk of damage to hardware. When satisfactory qualification tests have been conducted on a representative structural model, the strength qualification testing of the protoflight unit may not be necessary.

a. Selection of Test Method - The qualification load conditions may be applied by acceleration testing, static load testing, or vibration testing (either transient, fixed frequency or swept sinusoidal excitation). Random vibration is generally not acceptable for loads testing.

Consideration shall be given to the following when the method to be employed for verification tests is selected:

(1) The method most closely approximates the flight-imposed load distribution

(2) Application of flight load distribution with the greatest accuracy

(3) Method for deriving information for design verification and for predicting design capability for future observatory or launch vehicle modifications

(4) Posing the least risk to the hardware in terms of handling and test equipment

(5) Remaining within cost, time, and facility limitations

b. Test Setup - The subsystem/instrument shall be attached to the test equipment by a fixture whose mechanical interface simulates the mounting of the subsystem/instrument into the observatory with particular attention paid to duplicating the actual mounting contact area. In mating the subsystem to the fixture, a flight-type mounting (including vibration isolators or kinematic mounts if part of the design) and fasteners shall be used.

Components that are normally sealed shall be pressurized during the test to their prelaunch pressure. In cases when significant changes in strength, stiffness, or applied load result from variations in internal and external pressure during the launch phase, a special test shall be considered to cover those effects.

When acceleration testing is performed, the centrifuge shall be large enough so that the applied load at the extreme ends of the test item does not differ by more than 10 percent from that applied to the center of gravity. In addition, when the proper orientation for the applied acceleration vector is computed, ambient gravity effects shall be considered.

c. Performance - Before and after the strength qualification test, the subsystem/ instrument shall be examined and functionally tested to verify compliance with all performance criteria. During the tests, performance shall be monitored in accordance with the verification specification and procedures.

If appropriate development tests are performed to verify accuracy of the stress model, stringent quality control procedures are invoked to ensure conformance of the structure (materials, fasteners, welds, processes, etc.) to the design, and the structure has well-defined load paths, then strength qualification may (with LDCM project concurrence) be accomplished by a stress analysis that demonstrates that the hardware has positive margins on yield at loads equal to 2.0 times the limit load, and positive margin on ultimate at loads equal to 2.6 times the limit load. If warranted, factors of safety lower than 2.0 on yield and 2.6 on ultimate will be considered for approval by the GSFC LDCM project office. Justification for the lower factors of safety shall be based on the merits of a particular combination of test and analysis and a correlation of the two. Such alternative approaches shall be reviewed and approved on a case-by-case basis. In addition, at stresses equal to the limit load, the analysis shall show that the maximum allowable loads at the launch vehicle interface points are not exceeded and that no excessive deformations occur.

Structural elements fabricated from composite materials or beryllium shall not be qualified by analysis alone.

2.4.1.3.1 Strength Qualification - Beryllium - All beryllium primary and secondary structural elements shall undergo a strength test to 1.4 times limit load. No detrimental permanent deformation shall be allowed to occur as a result of applying the loads, and applicable alignment requirements shall be met following the test. In addition:

a. When using cross-rolled sheet, the design shall preclude out-of-plane loads and displacements during assembly, testing, or service life.

b. In order to account for uncertainties in material properties and local stress levels, a design factor of safety of 1.6 on ultimate material strength shall be used.

c. Stress analysis shall properly account for the lack of ductility of the material by rigorous treatment of applied loads, boundary conditions, assembly stresses, stress concentrations, thermal cycling, and possible material anisotropy. The stress analysis shall take into account worst-case tolerance conditions.

d. All machined and/or mechanically disturbed surfaces shall be chemically milled to ensure removal of surface damage and residual stresses.

e. All parts shall undergo penetrant inspection for surface cracks and crack-like flaws per MIL-STD-6866.

2.4.1.4 Structural Reliability (Residual Strength Verification) - Structural reliability requirements are intended to provide a high probability of the structural integrity of all flight hardware. They are generally covered by the selection of materials, process controls, selected analyses (stress, and fracture mechanics/crack growth), and loads/proof tests.

All structural materials contain defects such as inclusions, porosity, and cracks. To ensure that adequate residual strength (strength remaining after the flaws are accounted for) is present for structural reliability at launch, a fracture control program, or a combination of fracture control and specific loads tests, shall be performed on all flight hardware as specified below.

The use of materials that are susceptible to brittle fracture or stress-corrosion cracking require development of, and strict adherence to, special procedures to prevent problems. If materials are used for structural application that are not listed in Table 1 of MSFC-SPEC-522, a Materials Usage Agreement (MUA) shall be negotiated with the project office. Refer to project Materials and Processes Control Requirements for applicable requirements.

2.4.1.4.1 Primary and Secondary Structure:

ELV Payloads - The following requirements regarding beryllium, nonmetallic-composite, and metallic-honeycomb structural elements (both primary and secondary), and bonded structural joints apply to ELV payloads:

a. Beryllium Primary and Secondary Structure: The requirements of section 2.4.1.3.1, Strength Verification-Beryllium, apply for structural reliability.

b. Nonmetallic Composite Structural Elements (including metal matrix): It is preferred that all flight structural elements shall be proof tested to 1.25 times limit load (even if previously qualified on valid prototype hardware). However, if this is not feasible then it is acceptable to proof test a representative set of structural elements to 1.25 times the highest limit load for that type of structure. The remainder of the structural elements may then be considered qualified by similarity. In order to use this approach, the allowables used to assess structural margins shall be developed based on coupon testing and standard statistical techniques. As a minimum, B-basis allowables shall be used. In addition:

(1) A process control plan shall be developed and implemented to ensure uniformity of processing among test coupons, test articles, and flight hardware as required by the project Materials and Processes Control Requirements.

(2) A damage control plan shall be implemented to establish procedures and controls to prevent and/or identify nonvisible impact damage which may cause premature failure of composite elements.

c. Metallic Honeycomb (both facesheets and core) Structural Elements:

(1) Appropriate process controls and coupon testing shall be implemented to demonstrate that the honeycomb structure is acceptable for use as observatory flight structure as required by the project Materials and Processes Control Requirements.

(2) Metallic honeycomb is not considered to be a composite material.

d. Bonded Structural Joints (either metal-metal or metal-nonmetal):

(1) It is preferred that every bonded structural joint in a flight article shall be proof tested (by static loads test) to 1.25 times limit load. For example, proof loads testing shall be performed to demonstrate that inserts will not tear out from honeycomb under protoflight loads. However, in cases where this approach is not feasible, it is acceptable to test a representative sample of the bonded structural joints in the flight article. As a minimum, at least one of each type of bonded joint in the flight article shall be tested to 1.25 times the maximum predicted limit load for that joint type. The remainder of the bonded joints may then be considered to be qualified by similarity. The use of this approach requires that bonded joint allowables be developed based on coupon testing or testing of sample joints and standard statistical techniques. As a minimum, B-basis allowables shall be used.

(2) A process control plan shall be developed and implemented as required by applicable project Materials and Processes Control Requirements to ensure uniformity of processing among test coupons, test articles, and flight hardware.

ELV Payloads - If the observatory is to be placed in orbit by an ELV, fracture control requirements (per GSFC 731-0005-83) shall apply to the following elements only:

a. Pressure vessels, dewars, lines, and fittings (per NHB-8071.1),

b. Castings (unless hot isostatically pressed and the flight article is proof tested to 1.25 times limit load),

c. Weldments,

d. Parts made of materials on Tables II or III of MSFC-SPEC-522B if under sustained tensile stress. (Note: All structural applications of these materials requires that a Materials Usage Agreement (MUA) shall be negotiated with the project office; refer to project Materials and Processes Control Requirements,

e. Parts made of materials susceptible to cracking during quenching,

f. Nonredundant, mission-critical preloaded springs loaded to greater than 25 percent of ultimate strength.

All glass elements, that are stressed above 10% of their ultimate tensile strength, shall also be shown by fracture analysis to satisfy "Safe-life" or "Fail-safe" conditions or be subjected to a proof loads test at 1.0 times limit level.

2.4.1.5 Acceptance Requirements - All of the structural reliability requirements of 2.4.1.4 (as specified for ELV payloads) apply for the acceptance of all flight hardware.

Generally, structural design loads testing is not required for flight structure that has been previously qualified for the current mission as part of a valid prototype or protoflight test. However, the following acceptance/proof loads tests are required unless equivalent load-level testing was performed on the actual flight hardware as part of a protoflight test program:

a. For ELV Payloads

(1) Beryllium structure (primary and secondary) shall be proof tested to 1.4 times limit load.

(2) Nonmetallic composites (including metal matrix) structural elements shall be proof tested to 1.25 times limit load.

(3) Bonded structural joints shall be proof tested (by static loads test) to 1.25 times limit load.

If a follow-on observatory receives structural modifications or a new complement of instruments, it shall be requalified for the loads environment if analysis so indicates.

2.4.2 Vibroacoustic Qualification

Qualification for the vibroacoustics environment generally requires an acoustics test at the observatory level of assembly and random vibration tests on all components, instruments, and on the observatory, when appropriate, to better simulate the structure borne inputs. In addition, random vibration tests shall be performed on all subsystems unless an assessment of the expected environment indicates that the subsystem will not be exposed to any significant vibration input. Similarly, an acoustic test shall be performed on subsystems/instruments and components unless an assessment of the hardware indicates that they are not susceptible to the expected acoustic environment or that testing at higher levels of assembly provides sufficient exposure at an acceptable level of risk to the program. Irrespective of the above stated conditions, these additional tests may be required to satisfy delivery requirements.

It is understood that for some observatory projects, the vibroacoustic qualification program may have to be modified. For example, for very large observatories it may be impracticable because of test facility limitations to perform testing at the required level of assembly. In that case, testing at the highest practicable level of assembly shall be performed, and additional tests and/or analyses added to the verification program per approval by the GSFC LDCM project office. Also, the risk to the program associated with the modified test program shall be assessed and documented in the System Verification Plan.

Similarly, for very large components, the random vibration tests may have to be supplemented or replaced by an acoustic test. If the component level tests are not capable of inducing sufficient excitation to internal electric, electronic, and electromechanical devices to provide adequate workmanship verification, it is recommended that an environmental stress screening test program be conducted at lower levels of assembly (subassembly or board level).

For the vibroacoustic environment, limit levels shall be used which are consistent with the minimum probability levels of Table 2.4-2. The protoflight qualification level is defined as the flight limit level plus 3 dB. When random vibration levels are determined, responses to the acoustic inputs plus the effects of vibration transmitted through the structure shall be considered.

The random vibration test levels to be used for hardware containing delicate optics, sensors/detectors, etc., may be notched in frequency bands known to be destructive to the hardware with project concurrence. A force-limiting control strategy is recommended. This requires a dual control system which will automatically notch the input so as not to exceed design/expected forces in the area of rigid, shaker mounted resonances while maintaining acceleration control over the remainder of the frequency band. The control methodology shall be approved by the GSFC LDCM project office. More information on implementing the force-limiting control strategy can be found in Force Limited Vibration Testing NASA Technical Handbook, NASA-HDBK-7004.

As a minimum, the vibroacoustic test levels shall be sufficient to demonstrate acceptable workmanship.

During test, the test item shall be in an operational configuration, both electrically and mechanically, representative of its configuration at lift-off.

The vibroacoustic (acoustics plus random vibration) environmental test program shall be included in the environmental verification plan and environmental verification specification, which are reviewed by the GSFC Office of Mission Success.

2.4.2.1 Fatigue Life Considerations - The nature of the protoflight test program prevents a demonstration of hardware lifetime because the same hardware is both tested and flown. When hardware reliability considerations demand the demonstration of a specific hardware lifetime, a prototype verification program shall be employed, and the test durations shall be modified accordingly.

Specifically, the duration of the vibroacoustic exposures shall be extended to account for the life that the flight hardware will experience during its mission. In order to account for the scatter factor associated with the demonstration of fatigue life, the duration of prototype exposures shall be at least four times the intended life of the flight hardware. For ELV observatories, the duration of the exposure shall be based on both the vibroacoustic and sine vibration environments.

If there is the possibility of thermally induced structural fatigue (examples include solar arrays, antennas, etc.), thermal cycle testing shall be performed on prototype hardware. For large solar arrays, a representative smaller qualification panel may be used for test provided that it contains all of the full scale design details (including at least 100 solar cells) susceptible to thermal fatigue. The life test shall normally be performed at the worst case (limit level) predicted temperature extremes for a number of thermal cycles corresponding to the required mission life. However, if required by schedule considerations, the test program may be accelerated by increasing the temperature cycle range (and possibly the temperature transition rate) provided that stress analysis shows no unrealistic failure modes are produced by the accelerated testing.

2.4.2.2 Observatory Acoustic Test - At the observatory level of assembly, protoflight hardware shall be subjected to an acoustic test in a sound pressure field to verify its ability to survive the lift-off acoustic environment and to provide a final workmanship acoustic test. The test specification is dependent on the observatory-launch vehicle configuration and shallbe determined on a case-by-case basis and approved by the GSFC LDCM project office. The minimum overall test level shall be at least 138 dB. If the test specification derived from the launch vehicle expected environment, including fill-factor, is less than 138 dB, the test profile shall be raised to provide a 138 dB test level. The planned test and specification levels shall be confirmed by the launch vehicle program office.

a. Facilities and Test Control - The acoustic test shall be conducted in an area large enough to maintain a uniform sound field at all points surrounding the test item. The sound pressure level is controlled at one-third octave band resolution. The preferred method of control is to average four or more microphones with a real-time device that effectively averages the sound pressure level in each filter band. When real-time averaging is not practicable, a survey of the chamber shall be performed to determine the single point that is most suitable for control of the acoustic test.

Regardless of the control method employed, a minimum of four microphones shall be positioned around the test chamber at sufficient distance from all surfaces to avoid absorption or re-radiation effects. One of the microphones shall be located above the test item for a free-field test. A distance from any surface of at least l/4 the wavelength of the lowest frequency of interest is recommended. It is recognized that this cannot be achieved in some facilities, particularly when noise levels are specified to frequencies as low as 25 Hz. In such cases, the microphones shall be located in positions so as to be affected as little as possible by surface effects.

The preferred method of preparing for an acoustic test is to preshape the spectrum of the acoustic field with a dummy test item. If no such item is readily available, it is possible to preshape the spectrum in an empty test area. In that case, however, a low-level test shall be performed after the test item has been placed in the test area to permit final adjustments to the shape of the acoustic spectrum.

b. Test Setup - The boundary conditions under which the hardware is supported during test shall duplicate those expected during flight. When that is not feasible, the test item shall be mounted in the test chamber in such a manner as to be isolated from all energy inputs on a soft suspension system (natural frequency less than 20 Hz) and a sufficient distance from chamber surfaces to minimize surface effects. During test, the test item shall be in an operational configuration, both electrically and mechanically, representative of its configuration at lift-off.

c. Performance - Before and after the acoustic exposure, the observatory shall be examined and functionally tested. During the test, performance shall be monitored in accordance with the verification specification.

2.4.2.3 Observatory Random Vibration Tests - At the observatory level of assembly, protoflight hardware shall, when practicable, be subjected to a random vibration test to verify its ability to survive the lift-off environment and also to provide a final workmanship vibration test. For small observatories (<454 kg or 1000 lb), the test is required; for larger observatories the need to perform a random vibration test shall be assessed on a case-by-case basis and approved by the GSFC LDCM project office. Additional qualification tests may be required if expected environments are not enveloped by this test. The acoustic environment at lift-off is usually the primary source of random vibration; however, all other sources of random vibration shall be considered. The sources include transonic aerodynamic fluctuating pressures and the firing of retro/apogee motors.

a. Lift-Off Random Vibration - Protoflight hardware shall be subjected to a random vibration test to verify flightworthiness and workmanship. The test level shall represent the qualification level (flight limit level plus 3 dB).

The test shall cover the full 20-2000 Hz frequency range. Both lift-off and transonic random vibration shall be considered.

The observatory in its launch configuration shall be attached to a vibration fixture by use of a flight-type launch-vehicle adapter and attachment hardware. Vibration shall be applied at the base of the adapter in each of three orthogonal axes, one of which is parallel to the thrust axis. The excitation spectrum as measured by the control accelerometer(s) shall be equalized such that the acceleration spectral density is maintained within ± 3 dB of the specified level at all frequencies within the test range and the overall RMS level is within 10% of the specified level.

Prior to the observatory test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. If a mechanical test model of the observatory is available it shall be included in the survey to evaluate the need for limiting.

If a random vibration test is not performed at the observatory level of assembly, the feasibility of doing the test at the next lower level of assembly shall be assessed by the Contractor and approved by the GSFC LDCM project office.

b. Performance - Before and after each vibration test, the observatory shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

2.4.2.4 Subsystem/Instrument Vibroacoustic Tests - If subsystems are expected to be significantly excited by structure-borne random vibration, a random vibration test shall be performed. Specific test levels are determined on a case-by-case basis and approved by the GSFC LDCM project office. The levels shall be equal to the qualification level as predicted at the location where the input will be controlled. Subsystem acoustic tests may also be required if the subsystem is judged to be sensitive to this environment or if it is necessary to meet delivery specifications. A random vibration test is generally required for instruments.

2.4.2.5 Component/Unit Vibroacoustic Tests - As a screen for design and workmanship defects, components/units shall be subjected to a random vibration test along each of three mutually perpendicular axes. In addition, when components are particularly sensitive to the acoustic environment, an acoustic test shall be considered.

a. Random Vibration - The test item shall be subjected to random vibration along each of three mutually perpendicular axes for one minute each. When possible, the component random vibration spectrum shall be based on levels measured at the component mounting locations during previous subsystem or observatory testing. When such measurements are not available, the levels shall be based on statistically estimated responses of similar components on similar structures or on analysis of the observatory. Actual measurements shall then be used if and when they become available. In the absence of any knowledge of the expected level, the generalized vibration test specification of Table 2.4-3 shall be used.

As a minimum, all components shall be subjected to the levels of Table 2.4-4, which represent a workmanship screening test. The minimum workmanship test levels are primarily intended for use on electrical, electronic, and electromechanical hardware.

The test item shall be attached to the test equipment by a rigid fixture. The mounting shall simulate, insofar as practicable, the actual mounting of the item in the observatory with particular attention given to duplicating the mounting contact area. In mating the test item to the fixture, a flight-type mounting (including vibration isolators or kinematic mounts, if part of the design) and fasteners shall be used. Normally sealed items shall be pressurized during test to their prelaunch pressure.

In cases where significant changes in strength, stiffness, or applied load result from variations in internal and external pressure during the launch phase, a special test shall be considered to cover those effects.

Prior to the test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. The evaluation shall include consideration of cross-axis responses. If a mechanical test or engineering model of the test article is available it shall be included in the survey.

For very large components the random vibration tests may have to be supplemented or replaced by an acoustic test if the vibration test levels are insufficient to excite internal hardware. If neither the acoustic nor vibration excitation is sufficient to provide an adequate workmanship test, a screening program shall be initiated at lower levels of assembly; down to the board level, if necessary. The need for the screening program shall be evaluated by the project. The evaluation is based on mission reliability requirements and hardware criticality, as well as budgetary and schedule constraints.

If testing is performed below the component level of assembly, the workmanship test levels of Table 2.4-4 can be used as a starting point for test tailoring. The intent of testing at this level of assembly is to uncover design and workmanship flaws. The test input levels do not represent expected environments, but are intended to induce failure in weak parts and to expose workmanship errors. The susceptibility of the test item to vibration shall be evaluated and the test level tailored so as not to induce unnecessary failures.

If the test levels create conditions that exceed appropriate design safety margins or cause unrealistic modes of failure, the input spectrum can be notched below the minimum workmanship level. This can be accomplished when flight or test responses at the higher level of assembly are known or when appropriate force limits have been calculated.

b. Acoustic Test - If a component-level acoustic test is required, the test set-up and control shall be in accordance with the requirements for observatory testing.

c. Performance - Before and after test exposure, the test item shall be examined and functionally tested. During the test, performance shall be monitored in accordance with the verification specification.

2.4.2.6 Acceptance Requirements - Vibroacoustic testing for the acceptance of previously qualified hardware shall be conducted at flight limit levels using the same duration as recommended for protoflight hardware. As a minimum, the acoustic test level shall be 138 dB, and the random vibration levels shall represent the workmanship test levels.

The observatory is subjected to an acoustic test and/or a random vibration test in three axes. Components shall be subjected to random vibration tests in the three axes. Additional vibroacoustic tests at subsystem/instrument and component levels of assembly are performed in accordance with the environmental verification plan or as required for delivery.

During the test, performance shall be monitored in accordance with the verification specification.

Table 2.4-3
Generalized Random Vibration Test Levels
Components (ELV)
22.7-kg (50-lb) or less

Frequency (Hz)	ASD Level (g^2/Hz)	
	Qualification	Acceptance
20	0.026	0.013
20-50	+6 dB/oct	+6 dB/oct
50-800	0.16	0.08
800-2000	-6 dB/oct	-6 dB/oct
2000	0.026	0.013
Overall	14.1 G_{rms}	10.0 G_{rms}

The acceleration spectral density level may be reduced for components weighing more than 22.7-kg (50 lb) according to:

	Weight in kg	Weight in lb	
dB reduction	= 10 log(W/22.7)	10 log(W/50)	
$ASD_{(50\text{-}800\ Hz)}$	= 0.16•(22.7/W)	0.16•(50/W)	for protoflight
$ASD_{(50\text{-}800\ Hz)}$	= 0.08•(22.7/W)	0.08•(50/W)	for acceptance

Where W = component weight.

The slopes shall be maintained at + and - 6dB/oct for components weighing up to 59-kg (130-lb). Above that weight, the slopes shall be adjusted to maintain an ASD level of 0.01 g^2/Hz at 20 and 2000 Hz.

For components weighing over 182-kg (400-lb), the test specification will be maintained at the level for 182-kg (400 pounds).

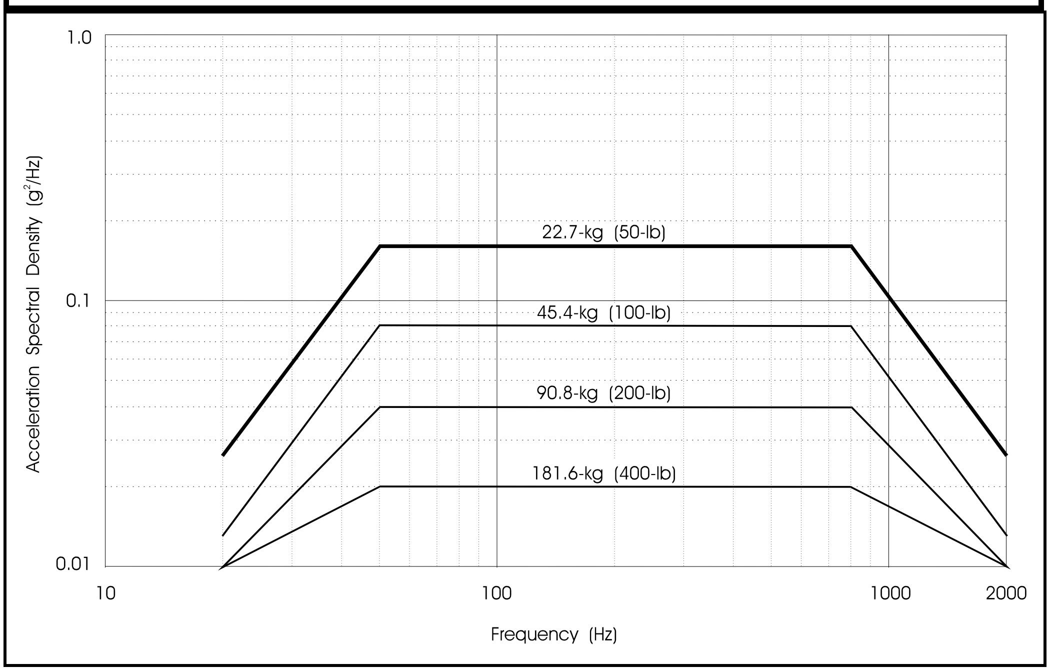


Table 2.4-4
Component Minimum Workmanship
Random Vibration Test Levels
45.4-kg (100-lb) or less

Frequency (Hz)	ASD Level (g^2/Hz)
20	0.01
20-80	+3 dB/oct
80-500	0.04
500-2000	-3 dB/oct
2000	0.01
Overall	6.8 g_{rms}

The plateau acceleration spectral density level (ASD) may be reduced for components weighing between 45.4 and 182 kg, or 100 and 400 pounds according to the component weight (W) up to a maximum of 6 dB as follows:

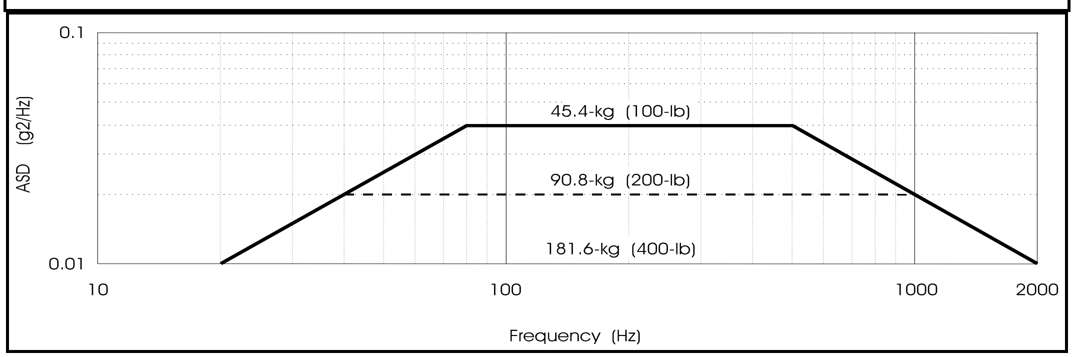
		Weight in kg	Weight in lb
dB reduction	=	10 log(W/45.4)	10 log(W/100)
$ASD_{(plateau)}$ level	=	0.04•(45.4/W)	0.04•(100/W)

The sloped portions of the spectrum shall be maintained at plus and minus 3 dB/oct. Therefore, the lower and upper break points, or frequencies at the ends of the plateau become:

F_L = 80 (45.4/W) [kg] F_L = frequency break point low end of plateau
= 80 (100/W) [lb]

F_H = 500 (W/45.4) [kg] F_H = frequency break point high end of plateau
= 500 (W/100) [lb]

The test spectrum shall not go below 0.01 g^2/Hz. For components whose weight is greater than 182-kg or 400 pounds, the workmanship test spectrum is 0.01 g^2/Hz from 20 to 2000 Hz with an overall level of 4.4 g_{rms}.



2.4.2.7 (Deleted)

2.4.2.8 Retest of Reworked Hardware – In many cases it is necessary to make modifications to hardware after a unit has been through a complete mechanical verification program. For example, replacing a capacitor on a circuit board in a electronics box that has already been through protoflight vibration testing. For this type of reworked hardware, the amount of additional mechanical testing required depends on the amount of rework done and the amount of disassembly performed as part of the rework. The primary objective of post-rework testing is to ensure proper workmanship has been achieved in performing the rework and in reassembling the component. As a minimum, the reworked component shall be subjected to a single axis workmanship random vibration test to the levels specified in Table 2.4-4. The determination of axis shall be made based on the direction necessary to provide the highest excitation of the reworked area. Testing may be required in more than one axis if a single axis test cannot be shown to adequately test all of the reworked area. If the amount of rework or disassembly required is significant, then 3-axis testing to acceptance levels may be necessary if they are higher than workmanship levels as deemed necessary by the MRB.

2.4.3 Sinusoidal Sweep Vibration Qualification

Sine sweep vibration tests are performed to qualify prototype/protoflight hardware for the low-frequency transient or sustained sine environments when they are present in flight, and to provide a workmanship test for all observatory hardware which is exposed to such environments and normally does not respond significantly to the vibroacoustic environment at frequencies below 50 Hz, such as wiring harnesses and stowed appendages.

Each observatory shall be assessed for such applicable sine test requirements. Qualification for these environments requires swept sine vibration tests at the observatory, instrument, and component levels of assembly. Test levels shall be developed on a mission-specific basis as addressed in 2.4.3.1 and 2.4.3.2.

For a observatory level test, the observatory shall be in a configuration representative of the time the stress occurs during flight, with appropriate flight type hardware used for attachment.

ELV observatories shall be subjected to swept sine vibration testing to simulate low-frequency sine transient vibration and sustained, pogo-like sine vibration (if expected) induced by the launch vehicle. Qualification for these environments requires swept sine vibration tests at the observatory, instrument, and component levels of assembly.

It is understood that, for some observatory projects, the sinusoidal sweep vibration qualification program may have to be modified. For example, for very large ELV observatories (with very large masses, extreme lengths, or large c.g. offsets) it may be impracticable because of test facility limitations to perform a swept sine vibration test at the observatory level of assembly. In that case, testing at the highest level of assembly practicable is required.

For the sinusoidal vibration environment, limit levels shall be used which are consistent with the minimum probability level given in Table 2.4-2. The qualification level is then defined as the limit level times 1.25. The test input frequency range shall be limited to the band from 5 to 50 Hz. The fatigue life considerations of 2.4.2.1 apply where hardware reliability goals demand the demonstration of a specific hardware lifetime. The sine sweep environmental test program shall be included in the environmental verification plan and environmental verification specification.

2.4.3.1 ELV Observatory Sine Sweep Vibration Tests - At the observatory level of assembly, prototype/protoflight hardware shall, when practicable, be subjected to a sine sweep vibration design qualification test to verify its ability to survive the low-frequency launch environment. The test also provides a workmanship vibration test for observatory hardware which normally does not respond significantly to the vibroacoustic environment at frequencies below 50 Hz, but can experience significant responses from the ELV low-frequency sine transient vibration and any sustained, pogo-like sine vibration. Guidelines for developing mission-specific test levels are given in 2.4.3.1.b.

a. Vibration Test Requirements - Protoflight hardware shall be subjected to a sine sweep vibration test to verify flightworthiness and workmanship. The test shall represent the qualification level (flight limit level times 1.25).

The test is intended for all ELV observatories (spacecraft) except those with very large masses, extreme lengths and/or large c.g. offsets, where it is impracticable because of test facility limitations.

If the sine sweep vibration test is not performed at the observatory level of assembly, it shall be performed at the next lowest practicable level of assembly.

The observatory in its launch configuration shall be attached to a vibration fixture by use of a flight-type launch-vehicle attach fitting (adapter) and attachment (separation system) hardware. Sine sweep vibration shall be applied at the base of the adapter in each of three orthogonal axes, one of which is parallel to the thrust axis. The test sweep rate shall be 4 octaves per minute to simulate the flight sine transient vibration; lower sweep rates shall be used in the appropriate frequency bands as required to match the duration and rate of change of frequency of any flight sustained, pogo-like vibration. The test shall be performed by sweeping the applied vibration once through the 5 to 50 Hz frequency range in each test axis. Mission-specific sine sweep test levels shall be developed for each ELV payload. Guidelines for developing the test levels are given in 2.4.3.1.b.

Prior to the observatory test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. The evaluation shall include consideration of cross-axis responses. If a mechanical test model of the observatory is available it shall be included in the survey to evaluate the need for limiting (or notching).

During the protoflight hardware sine sweep vibration test to the specified test levels, loads induced in the observatory and/or adapter structure while sweeping through resonance shall not exceed 1.25 times flight limit loads. If required, test levels shall be reduced ("notched") at critical frequencies. Acceleration responses of specific critical items may also be limited to 1.25 times flight limit levels if required to preclude unrealistic levels, provided that the observatory model used for the coupled loads analysis has sufficient detail and that the specific responses are recovered (using the acceleration transformation matrix) from the coupled loads analysis results. The minimum controlled input test level shall be 0.1 g to facilitate shaker control.

A low-level sine sweep shall be performed prior to the protoflight-level sine sweep test in each test axis. Data from the low-level sweeps measured at locations identified by a notching analysis shall be examined to determine if there are any significant test response deviations from analytical predictions. The data utilized shall include cross-axis response levels. Based on the results of the low-level tests, the predetermined notch levels shall be verified prior to the protoflight-level test. The flight limit loads used for notching analysis shall be based on the final verification cycle coupled loads analysis (including a test-verified observatory model).

b. Mission-Specific Test Level Development - Sinusoidal vibration test levels required to simulate the flight environment for ELV observatory vary with the observatory attach fitting (adapter) and observatory configuration, including overall weight and length, mass and stiffness distributions, and axial-to-lateral coupling. It therefore is impracticable to specify generalized sine sweep vibration test levels applicable to all observatory, and mission-specific test levels shall be developed for each ELV observatory based on the coupled loads analysis.

Prior to the availability of coupled loads analysis results, preliminary sine test levels may be estimated by using the ELV "user manual" sine vibration levels for observatory base drive analysis, with notching levels based on net loads equivalent to the user manual c.g. load factor loads. Alternatively, observatory interface dynamic response data from flight measurements or coupled loads analysis for similar observatory may be used for the base drive input in conjunction with a suitable uncertainty factor approved by the GSFC LDCM project office.

c. Performance - Before and after each vibration test, the observatory shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

2.4.3.2 ELV Payload Subsystem (including Instruments) and Component Sine Sweep Vibration Tests - As a screen for design and workmanship defects, these items (per Table 2.4-1) shall be subjected to a sine sweep vibration test along each of three mutually perpendicular axes. For the sinusoidal vibration environment, limit levels shall be defined to be consistent with the minimum probability level of Table 2.4-2. The protoflight qualification level is then defined as the limit level times 1.25. The test input frequency range shall be limited to the band from 5 to 50 Hz. The fatigue life considerations of 2.4.2.1 apply where hardware reliability goals demand the demonstration of a specific hardware lifetime.

a. Vibration Test Requirements - The test item in its launch configuration shall be attached to the test equipment by a rigid fixture. The mounting shall simulate, insofar as practicable, the actual mounting of the item in the observatory, with particular attention given to duplicating the mounting interface. All connections to the item (connectors and harnesses, plumbing, etc.) shall be simulated with lengths at least to the first tie-down point. In mating the test item to the fixture, a flight-type mounting (including vibration isolators or kinematic mounts, if part of the design) and fasteners, including torque levels and locking features, shall be used. Normally-sealed items shall be pressurized during test to their prelaunch pressure.

In cases where significant changes in strength, stiffness, or applied load result from variations in internal and external pressure during the launch phase, a special test shall be considered to cover those effects.

Sine sweep vibration shall be applied at the base of the test item in each of three mutually perpendicular axes. The test sweep rate shall be consistent with the observatory-level sweep rate, i.e., 4 octaves per minute to simulate the flight sine transient vibration, and (if required) lower sweep rates in the appropriate frequency bands to match the duration and rate of change of frequency of any flight sustained, pogo-like vibration. The test shall be performed by sweeping the applied vibration once through the 5 to 50 Hz frequency range in each test axis.

Observatory subsystems, including instrument, and component levels depend on the type of structure to which the item is attached, the local attachment stiffness, the distance from the observatory separation plane, and the item's mass, size, and stiffness. It therefore is impracticable to specify generalized sine sweep vibration test levels applicable to all subsystems/instruments, and components, and mission-specific test levels shall be developed for each observatory. Guidelines for developing the specific test levels are given in 2.4.3.2.b.

Prior to the test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. The evaluation shall include consideration of cross-axis responses. If a mechanical test or engineering model of the test article is available it shall be included in the survey.

A low-level sine sweep shall be performed prior to the protoflight level sine sweep test in each test axis (with particular emphasis on cross-axis responses) to verify the control strategy and check test fixture dynamics.

b. Mission Specific Test Level Development - The mission-specific sine sweep test levels for observatory subsystems/components shall be based on test data from structural model observatory sine sweep tests if available. If not available, the test levels shall be based on an envelope of two sets of responses:

(1) Coupled loads analysis dynamic responses shall be utilized if acceleration-response time histories are available at the test article location for all significant flight event loading conditions. Equivalent sine sweep vibration test input levels shall be developed using shock response spectra (SRS) techniques for transient flight events. It should be noted that, in developing equivalent test input levels by dividing the SRS by Q (where $Q=C_C/2C$), assumption of a lower Q is more conservative. In the absence of test data, typical assumed values of Q for subsystems/components are from

10 to 20. For pogo-like flight events, the use of SRS techniques is not generally required.

(2) Subsystem/component responses from a base drive analysis of the observatory and adapter, using the observatory sine sweep test levels as input (in three axes), shall be included in the test level envelope. The base drive responses of the test article shall be corrected for effects of the observatory test sweep rates if the sweep rates are not included in the base drive analysis input. Subsystem/component test sweep rates shall match observatory test sweep rates.

If the facility's shaker can only apply translational (but not rotational) accelerations, then for test articles with predicted large rotational responses, the test levels shall be increased based on analysis to assure adequate response levels.

Also, for certain cases such as large items mounted on kinematic mount flexures, which experience both significant rotations and translations, it may be necessary to use the test article c.g. rotational and translational acceleration response levels as not-to-exceed test levels in conjunction with appropriate notching or limiting.

c. Performance - Before and after test exposure, the test item shall be examined and functionally tested. During the test, performance shall be monitored in accordance with the verification specification.

2.4.3.3 Acceptance Requirements - Sine sweep vibration testing for the acceptance of previously qualified hardware shall be conducted at the flight limit levels using the same sweep rates as used for protoflight hardware.

2.4.4 Mechanical Shock Qualification

Both self-induced and externally induced shocks shall be considered in defining the mechanical shock environment.

2.4.4.1 Subsystem Mechanical Shock Tests - All subsystems, including instruments, shall be qualified for the mechanical shock environment.

a. Self-Induced Shock - The subsystem shall be exposed to self-induced shocks by actuation of all shock-producing devices. Self-induced shocks occur principally when pyrotechnic and pneumatic devices are actuated to release booms, solar arrays, protective covers, etc. Also the impact on deployable devices as they reach their operational position at the "end of travel" is a likely source of significant shock. When hardware contains such devices, it shall be exposed to each shock source twice to account for the scatter associated with the actuation of the same device. The internal observatory flight firing circuits shall be used to trigger the event rather than external test firing circuits. With the GSFC LDCM project's approval, this testing may be deferred to the observatory level of assembly.

b. Externally Induced Shock - Mechanical shocks originating from other subsystems, observatories, or launch vehicle operations shall be assessed. When the most severe shock is externally induced, a suitable simulation of that shock shall be applied at the subsystem interface. When it is feasible to apply this shock with a controllable shock-generating device, the qualification level shall be 1.4 times the maximum expected value at the subsystem interface, applied once in each of the three axes. A pulse or complex transient (whose positive and negative shock spectrum matches the desired spectrum within +25% and -10%) with a duration of 10ms or less shall be applied to the test item interface once along each of the three axes. Equalization of the shock spectrum is performed at a maximum resolution of one-third octave. The fraction of critical damping (c/c_C) used in the shock spectral analysis of the test pulse shall equal the fraction of critical damping used in the analysis of the data from which the test specification was derived. In the absence of a strong rationale for some other value, a fraction of critical damping equivalent to a Q of 10 shall be used for shock spectrum analysis.

If the GSFC LDCM project approves that it is not feasible to apply the shock with a controllable shock-generating device (e.g. the subsystem is too large for the device), the test may be conducted at the observatory level by actuating the devices in the observatory that produce the shocks external to the subsystem to be tested. The shock-producing device(s) shall be actuated a minimum of two times for this test.

The decision to perform component shock testing to is typically based on an assessment of the shock susceptibility of the component and the expected shock levels. If there is low potential for damage due to the shock environment, then shock testing may be deferred with GSFC LDCM project approval to the observatory level of assembly. For standard electronics, the potential for damage due to shock can be quantified based on Figure 2.4-1 If the flight shock environment as shown on an SRS plot (Q=10) is enveloped by the curve shown in Figure 2.4-1, then the shock environment can be considered benign and there is low risk in deferring the shock test. For the case in which the shock levels are above the curve, then component level shock testing shall be considered. The curve provided in Figure 2.4-1 is intended as a guideline for determining whether component level shock testing should be performed. Each component should be evaluated individually to determine its susceptibility for damage due to the predicted shock environment.

It will not be necessary to conduct a test for externally induced shocks if it can be demonstrated that the shock spectrum of the self-induced environment is greater at allfrequencies than the envelope of the spectra created by the external events at all locations within the subsystem.

d. Test Setup - During test, the test item shall be in the electrical and mechanical operational modes appropriate to the phase of mission operations when the shock will occur.

e. Performance - Before and after the mechanical shock test, the test item shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

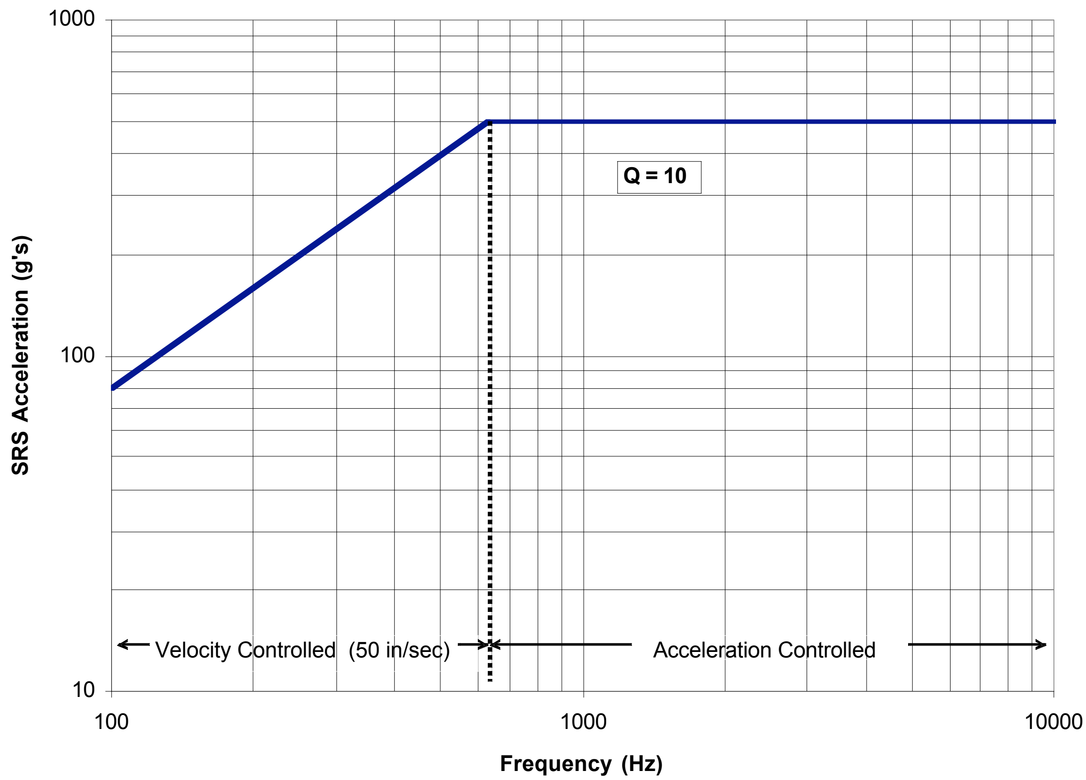


Figure 2.4-1 Shock Response Spectrum (SRS) for assessing Component Test Requirements

2.4.4.2 Observatory Mechanical Shock Tests - The observatory shall be qualified for the shock induced during payload separation (when applicable) and for any other externally induced shocks whose levels are not enveloped at the payload interface by the separation shock level. The payload separation shock is usually higher than other launch vehicle-induced shocks; however that is not always the case. For instance, the shocks induced at the payload interface during inertial upper stage (IUS) actuation can be greater. In addition, mechanical shock testing may be performed at the observatory level of assembly to satisfy the subsystem mechanical shock requirements of 2.4.4.1.

a. Other Observatory Shocks - If launch vehicle induced shocks or shocks from other sources are not enveloped by the separation test, the observatory shall be subjected to a test designed to simulate the greater environment. If a controllable source is used, the qualification level shall be 1.4 x the maximum expected level at the payload interface applied once in each of the three axes. The tolerance band on the simulated level of response is +25% and -10%. The analysis shall be performed with a fraction of critical damping corresponding to a Q of 10 or, if other than 10, with the Q for which the shock being simulated was analyzed.

The subsystem mechanical shock requirements may be satisfied by testing at the observatory level of assembly as described above.

b. Performance - Before and after the mechanical shock test, the test item shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification test plan and specification.

2.4.4.3 Acceptance Requirements - Elimination of mechanical shock tests for acceptance of previously qualified hardware shall be considered for approval by the GSFC LDCM project office. Testing shall be given careful consideration evaluating mission reliability goals, shock severity, hardware susceptibility, design changes from the previous qualification configuration including proximity to the shock source, and previous history.

2.4.5 Mechanical Function Verification

A kinematic analysis of all observatory mechanical operations shall be performed (a) to ensure that each mechanism can perform satisfactorily and has adequate margins under worst-case conditions, (b) to ensure that satisfactory clearances exist for both the stowed and operational configurations as well as during any mechanical operation, and (c) to ensure that all mechanical elements are capable of withstanding the worst-case loads that may be encountered. Observatory qualification tests shall be performed to demonstrate that the installation of each mechanical device is correct and that no problems exist that will prevent proper operation of the mechanism during mission life.

Subsystem qualification tests shall be performed for each mechanical operation at nominal-, low-, and high-energy levels. To establish that functioning is proper for normal operations, the nominal test shall be conducted under the most probable conditions expected during normal flight. A high-energy test and a low-energy test shall also be conducted to prove positive margins of strength and function. The levels of these tests shall demonstrate margins beyond the nominal conditions by considering adverse interaction of potential extremes of parameters such as temperature, friction, spring forces, stiffness of electrical cabling or thermal insulation, and, when applicable, spin rate. Parameters to be varied during the high- and low-energy tests shall include, to the maximum extent practicable, all those that could substantively affect the operation of the mechanism as determined by the results of analytic predictions or development tests. As a minimum, successful operation at temperature extremes 10∞C beyond the range of expected flight temperatures shall be demonstrated.

Lubricants susceptible to adverse affects from humidity, such as MoS2 shall be given protection where storage or non-operating environments can be expected to cause adverse effects. Testing in a humid environment shall, where practicable, either be avoided or minimized.

2.4.5.1 Life Testing

A life test program shall be implemented for mechanical elements that move repetitively as part of their normal function and whose useful life shall be determined in order to verify their adequacy for the mission. The verification plan and the verification specification shall address the life test program, identifying the mechanical elements that

require such testing, describing the test hardware that will be used, and the test methods that will be employed.
Life test planning shall be initiated as early as possible in the development phase, and presented at each program system/peer review to allow enough time to complete the life test and thoroughly disassemble and inspect the mechanism, while retaining enough time to react to any anomalous findings. Once the plan is finalized, an independent peer review of the procedure and criteria shall be held.

The life test mechanism shall be fabricated and assembled such that it is as nearly identical as possible to the actual flight mechanism, with special attention to the development and implementation of detailed assembly procedures and certification logs. In fact, it is preferable that the life test mechanism actually be a flight spare or Qualification Unit. Careful attention shall be given to properly simulating the flight interfaces, especially the perhaps less obvious details, such as the method of mounting of the mechanism, the preloading and/or clamping of bearings or other tribological interfaces, the routing of harnesses, the attachment of thermal blankets, and any other items that could have an influence on the performance of the mechanism.

Prior to the start of life testing, mechanisms shall be subjected to the same ground testing environments, both structural and thermal, that are anticipated for the flight units (protoflight or acceptance levels, as appropriate). These environments may have a significant influence on the life test performance of the mechanism.

Consideration shall be given to the geometry of the test set-up and the effects of gravity on the performance of the life test mechanism, including the effects on lubrication and external loads. For example, gravity may cause lubrication to puddle at the bottom of a bearing race or run out of the bearing. In some cases, the effects of gravity may cause abnormally high loads on the mechanism.

The thermal environment of the mechanism during the life test shall be representative of the on-orbit environment. If expected bulk temperature changes are significant, then the life test shall include a number of transitions from the hot on-orbit predictions to the cold on-orbit predictions, and vice versa. Depending on the thermal design, significant temperature gradients may be developed which could have a profound influence on the life of the mechanism and, therefore, shall be factored into the thermal profile for the life test.

Consideration shall be given to including in the life test the effects of vacuum on the performance of the mechanism with particular attention to its effects on the thermal environment (i.e., no convective heat transfer) and potentially adverse effects on lubrication and materials. Life testing in a gaseous nitrogen environment as an inexpensive alternative to a long duration vacuum test, for example, may have a completely unexpected or unanticipated affect on lubricant tribology.

Life testing of electrically powered devices shall be conducted with nominal supply voltage.

The selection of the proper instrumentation for the life test is very important. Physical parameters that are an indication of the health of the mechanism shall be closely monitored and trended during the life test. These parameters may include in-rush and steady-state currents, electrical opens or shorts, threshold voltages, temperatures (both steady-state and rate of change), torques, angular or linear positions, vibration, times of actuation and open/closed loop system responses.

The life test shall be designed to "fail safe" in the event of any failure of the test setup, ground support equipment, or test article. There may be a severe impact to the life test results if it is necessary to stop a life test to replace or repair ground support equipment. Uninterruptible power supplies shall be considered when required for autonomous shutdown without damage to the test article or loss of test data. Redundant sensors shall be provided for all critical test data. If used, the vacuum pumping station shall be designed to maintain the integrity of the vacuum in the event of a sudden loss of power. Any autonomous data capture shall include a time stamp to help diagnose the conditions present prior to a test shutdown.

The test spectrum for the life test shall represent the required mission life for the flight mechanism, including both ground and on-orbit mechanism operations. In order to reduce test time and cost, the test spectrum shall be simplified as much as possible while retaining an appropriate balance between realism and conservatism. It shall include, if applicable, a representative range of velocities, number of direction reversals, and number of dead times or stop/start sequences between movements. Direction reversals and stop/start operations could have a significant effect on lubrication life, internal stresses, and, ultimately, the long term performance of the mechanism and therefore shall be given priority in the development of the life test plan. Similarly, system dynamics effects due to inertial loads shall be considered in development of the plan and implemented where appropriate, such as in applications where normal operation includes multiple start / stop or acceleration / deceleration maneuvers.

The minimum requirement for demonstrated life test operation without failure shall be 2.0 times the mission life. However, due to the uncertainties and simplifications inherent in the test, a marginally successful test requires post-test inspections and characterizations to extrapolate the remaining useful life. Because this can be difficult and uncertain, even higher margins shall be considered if time permits in order to establish greater confidence due to the limited number of life test units that are typically available. Pre- and post-life test baseline performance tests shall be conducted with clear requirements established for determining minimum acceptable performance at end-of-life.

When it is necessary to accelerate the life test in order to achieve the required life demonstration in the time available, caution shall be exercised in increasing the speed or duty cycle of the mechanism. Mechanisms may survive a life test at a certain speed or duty cycle, but fail if the speed is increased or decreased, or if the duty cycle is increased significantly. There are three lubrication regimes to consider when considering whether to accelerate a life test, "boundary lubrication", "mixed lubrication", and "full elastohydrodynamic (EHD) lubrication".

For boundary and mixed lubrication regimes, the most likely failure mechanisms will be wear and lubricant breakdown, not fatigue. Unfortunately failure by wear is not an exact science; therefore, life test acceleration by increasing speed shall be considered with caution. A mechanism that normally operates in these two regimes shall never be accelerated in a life test to a level where the lubrication system moves into the EHD regime for the test. Acceleration of a life test for systems in boundary or mixed lubrication regimes may be considered if it can be shown by analysis or test that the mechanism rotor oscillations for the accelerated operation are similar to that during normal operation. For example, in a step motor, it shall be shown that the rotor oscillations damp out to less than 10% of the peak overshoot amplitude prior to initiating the next accelerated step. Rationale for acceleration shall be presented in the initial test plan.

In the EHD regime, no appreciable wear shall occur and the failure mechanism shall be material fatigue rather than wear. Therefore, while life test acceleration by increasing speed may be considered, other speed limiting factors shall also be considered. For example, at the speed at which EHD lubrication is attained, one shall be concerned with bearing retainer imbalances which may produce excessive wear of the retainer, which would in turn produce contaminants which could degrade the performance of the bearings. Additionally, thermal issues may arise related to increased power dissipation for higher speed operation, like increased bearing gradients, which shall be thoroughly evaluated.

If there are significant downtimes associated with the operation of an intermittent mechanism, the life test can be accelerated by reducing this downtime, as long as this does not adversely affect temperatures and leaves enough "settle time" for the lubricant film to "squish out" of the contact area to simulate a full stop condition.

For all these reasons, the life test shall be run as nearly as possible using the on-orbit speeds and duty cycles. In some cases it may not be possible to accelerate the test at all.

Upon completion of the life test, it is imperative that careful disassembly procedures are followed and that the proper level of inspections are conducted. Successful tests will not have any anomalous conditions such as abnormal wear, significant lubrication breakdown, or excessive debris generation. These or other anomalous conditions may be cause for declaring the life test a failure despite completion of the required test spectrum. A thorough investigation of all moving components and wear surfaces shall be conducted. This may include physical dimensional inspection of components, high magnification photography, lubricant analysis, Scanning Electron Microscope (SEM) analysis, etc. Photographic documentation of the life test article shall be made from incoming component inspection/acceptance through full assembly to act as a baseline for comparison.

For those mechanical elements that move repetitively as part of their normal function determined not to require life testing, the rationale for eliminating the test along with the analyses to verify the validity of the rationale shall be provided for approval by the GSFC LDCM project office. Reference Section 2.2.3.

2.4.5.2 Demonstration - Compliance with the mechanical function qualification requirements shall be demonstrated by a combination of analysis and test. The functional qualification aspects of the demonstration are discussed below. The life test demonstrations shall be described in detail in an approved verification plan and verification specification.

a. Analysis - An analysis of the observatory shall be conducted to ensure that satisfactory clearances exist for both the stowed and operational configurations. Therefore, in conjunction with the flight-loads analysis, an assessment of the relative displacements of the various observatory elements with respect to other observatories and various elements of the ELV payload fairing shall be made for potentially critical events. During analysis, the following effects shall be considered: an adverse build-up of tolerances, thermal distortions, and mechanical misalignments, as well as the effects of static and dynamic displacements induced by particular mission events.

In addition, a kinematic analysis of all deployment and retraction sequences shall be conducted to ensure that each mechanism has adequate torque margin under worst-case friction conditions and is capable of withstanding the worst-case loads that may be encountered during unlatching, deployment, retraction, relatching, or ejection sequences. In addition, the analysis shall verify that sufficient clearance exists during the motion of the mechanisms to avoid any interference.

The selection of lubricant for use in critical moving mechanical assemblies shall be based upon development tests of the lubricant that demonstrate its ability to provide adequate lubrication under all specified operating conditions over the design lifetime. Since life testing cannot typically provide proof of lubricant availability based on evaporation over the required life of the mechanism, an analysis shall be performed to show that there is an adequate amount of lubricant in the system (not including degradation) for the duration of the mechanism life with a margin greater than 10. Lubricant availability analyses based on degradation rates shall be proven through life testing (see section 2.4.5.1).

The design of each ball bearing installation shall be substantiated by analysis and either development tests or previous usage. The materials, stresses, stiffness, fatigue life, preload, and possible binding under normal, as well as the most severe combined loading conditions, and other expected environmental conditions shall be considered. Alignments, fits, tolerances, thermal and load induced distortions, and other conditions shall be considered in determining preload variations. Bearing fatigue life calculations shall be based on a survival probability of 99.95 percent when subjected to maximum time varying loads. For noncritical applications or deployables, if nonquiet running is acceptable, and the bearing material is 52100 Carbon Steel or 440C Stainless Steel, the mean Hertzian contact stress shall not exceed 2760 megapascals (400,000 psi) when subjected to the yield load. During operation, the mean Hertzian contact stress shall not exceed 2310 megapascals (335,000 psi). For materials other than these, a hertzian contact stress allowable shall be determined based on manufacturer recommendations with appropriate reduction factors for aerospace applications and approved by the GSFC LDCM project office. In addition to the requirements stated above, bearing applications requiring quiet operation or low torque ripple shall be designed so that the bearing race and ball stress levels are below the levels that would cause unacceptable permanent deformation during application of ascent loads. Where bearing deformation is required to carry a portion or all of the vehicle ascent loads, and where smoothness of operation is required on orbit, the mean Hertzian stress levels of the bearing steel (52100 and 440C) shall not exceed 2310 megapascals (335,000 psi) when subjected to the yield load. The upper and lower extremes of the contact ellipses shall be contained by the raceways. The stress and shoulder height requirements of the races shall be analyzed for both nominal and off-nominal bearing tolerances. During operation, the mean Hertzian contact stress shall not exceed 830 megapascals (120,000 psi) over the worst case environment. For materials other than 52100 carbon steel and 440C stainless steel, a hertzian contact stress allowable shall be determined based on manufacturer recommendations with appropriate reduction factors for aerospace applications and approved by the GSFC LDCM project office.

b. Observatory Testing - A series of mechanical function tests shall be performed on the observatory to demonstrate "freedom-of-motion" of all appendages and other mechanical devices whose operation may be affected by the process of integrating them with the observatory. The tests shall demonstrate proper release, motion, and lock-in of each device, as appropriate, in order to ensure that no tolerance buildup, assembly error, or other problem will prevent proper operation of the mechanism during mission life. Unless the design of the device dictates otherwise, mechanical testing may be conducted in ambient laboratory conditions. The testing shall be performed at an appropriate time in the observatory environmental test sequence and, if any device is subsequently removed from the observatory, the testing shall be repeated after final reinstallation of the device.

c. Subsystem Testing - Each subsystem, and instrument, that performs a mechanical operation shall undergo functional qualification testing. With the GSFC LDCM project office approval, such testing may be performed at the observatory level of assembly. The test is conducted after any other testing that may affect mechanical operation. The purpose is to confirm proper performance and to ensure that no degradation has occurred during the previous tests.

During the test, the electrical and mechanical components of the subsystem shall be in the appropriate operational mode. The subsystem shall also be exposed to pertinent environmental effects that may occur before and during mechanical operation. The verification specification shall stipulate the tests to be conducted, the necessary environmental conditioning, and the range of required operations.

(1) Information Requirements - The following information shall be provided to define the series of functional qualification tests:

- o A description of mission requirements, how the mechanism is intended to operate, and when operation occurs during the mission;
- o The required range of acceptable operation and criteria for acceptable performance;
- o The anticipated variation of all pertinent flight conditions or other parameters that may affect performance.

(2) Test Levels and Margins - For each mechanical operation, such as appendage deployment, tests at nominal-, low-, and high-energy levels shall be performed. One test shall be conducted at the most probable level that will occur during a normal mission (the nominal level). The test will establish that functioning is proper for nominal operating conditions and baseline measurements will be obtained for subsequent tests.

Other tests shall be conducted to prove positive margins of strength and function, including torque or force ratio, a high-energy test and a low-energy test. The levels of these tests shall demonstrate margins beyond the nominal operational limits over the full range of motion at the worst case environments and the operating parameters of the system (rate, acceleration, etc.). The margins shall not be selected arbitrarily, but shall take into account all the uncertainties of operation, strength, and test. If a margin test cannot be conducted at the subsystem level due to its size and complexity these verification tests shall be performed at the highest level of assembly possible and the results combined to provide subsystem performance.

While in an appropriate functional configuration the hardware shall be subjected to events such as separation, appendage deployment, retromotor ejection, or other mechanical operations, such as spin-up or despin that are associated with the particular mission.

Gravity compensation shall be provided to the extent necessary to achieve the test objectives. As a guide, the uncompensated gravity effects shall be less than 10 percent of the operational loads. Uncompensated gravity of

0.1 g is usually achievable and acceptable for separation tests and for comparative measurements of appendage positioning if the direction is correct, i.e., the net shear and moment imposed during measurements acts in the same direction as it would in flight, thereby causing any mechanism with backlash to assume the correct extreme positions. For testing of certain mechanical functions, however, more stringent uncompensated gravity constraints may be required. When appropriate, the subsystem shall be preconditioned before test or conditioned during test to pertinent environmental levels. This can include vibration, high- and low-temperature cycling, pressure-time profiles, transportation and handling.

(3) Performance - Before and after test, the subsystem shall be examined and electrically tested. During the test, the subsystem performance shall be monitored in accordance with the verification specification.

(4) Component Characterization and Testing – For applications where motor performance is critical to mission success, the design shall be based on a complete motor characterization at the minimum and maximum voltages from the spacecraft bus and motor driver and shall include as a minimum: rotor inertia, friction and damping parameters, back-EMF constant or torque constant, time constant, torque characteristics, speed versus torque curves, thermal dissipation, temperature effects, and where applicable, analysis to demonstrate adequate margin against back driving.

For applications where the motor is integrated into a higher assembly, the motor characterization shall be performed at the motor level prior to integration.

After initial functional testing, a run-in test shall be performed on each moving mechanical assembly before it is subjected to further acceptance testing, unless it can be shown that this procedure would be detrimental to performance and would result in reduced reliability. The primary purpose of the run-in test is to detect material and workmanship defects that occur early in the component life. Another purpose is to wear-in parts of the moving mechanical assembly so that they perform in a consistent and controlled manner. Satisfactory wear-in may be manifested by a reduction in running friction to a consistent low level. The run-in test shall be conducted for a minimum of 50 hours except for items where the number of cycles of operation, rather than hours of operation, is a more appropriate measure of the capability to perform in a consistent and controlled manner. For these units, the run-in test shall be for at least 15 cycles or 5% of the total expected life cycles, whichever is greater. The run-in test conditions shall be representative of the operational loads, speed, and environment; however, operation of the assembly at ambient conditions may be conducted if the test objectives can be met and the ambient environment will not degrade reliability or cause unacceptable changes to occur within the equipment such as generation of excessive debris. During the run-in test, sufficient periodic measurements shall be made to indicate what conditions may be changing with time and what wear rate characteristics exist. Test procedures, test time, and criteria for performance adequacy shall be in accordance with an approved test plan. All gear trains using solid or liquid lubricants shall, where practicable, be inspected and cleaned following the run-in test.

 SYSTEM PERFORMANCE VERIFICATION

2.4.5.3 Torque/Force Margin - The torque or force margin shall be demonstrated by test to be sufficiently large to guarantee system-performance under worst-case conditions throughout its life by fully accommodating the uncertainty in the resisting forces or torques and in the source of energy.

The Torque Margin (TM) is a measure of the degree to which the torque available to accomplish a mechanical function exceeds the torque required. The torque margin is generally the ratio of the driving or available torques times an appropriate Factor of Safety (FS) minus one.. The torque margin requirement defined below applies to all mechanical functions, those driven by motors as well as springs, etc. at beginning of life (BOL) only; end of life (EOL) mechanism performance is determined by life testing as discussed in paragraph 2.4.5.1, and/or by analysis; however, all torque increases due to life test results shall be included in the final TM calculation and verification. Positive margin (>0) using the TM equation and FS stated herein shall be shown for worst case EOL predicted conditions and at the extreme operating parameters of the system (rate, acceleration, etc.). For linear devices, the term "force" shall replace "torque" throughout the section.

For final design verification, the torque margin shall be verified by testing the qualification (or protoflight) unit both before and after exposure to qualification level environmental testing. The torque margin on all flight units shall also be verified by testing when possible (without breaking the flight hardware configuration), both before and after exposure to acceptance level environmental testing. All torque margin testing shall be performed at the highest possible level of assembly, throughout the mechanism's range of travel, under worst-case predicted EOL environmental conditions, representing the worst-case combination of maximum and/or minimum predicted (not qualification) temperatures, gradients, positions, acceleration/ deceleration of load, rate, voltage, vacuum, etc. As the deviation from these worst case conditions increases, a higher Factor of Safety than that stated below shall be used.

Along with system level test, available torque (T_{avail}) and resistive torque (T_r) under worst case conditions shall be determined, whenever possible, through component, system and subsystem level tests. Torque ratios for gear driven systems shall be verified, using subsystem level results, on both sides of the geartrain. The minimum available torque for these types of systems shall never be less than 1 in-oz at the motor. Kick-off springs that do not operate over the entire range of the mechanical function shall be neglected when computing available torque over the full range. However, the use of kick-off spring forces in the Torque Margin calculation at the beginning of travel or initial separation is acceptable. A Factor of Safety of at least 1.5 over inertial driven or known quantifiable resistive torques (that do not change over the operating life of the unit) shall be used in the final computing of torque margin as indicated in the table below. FS requirements for parasitic forces dominated by a combination of variable items shall be determined based on the program phase as indicated in the table below. See Section 2.2.3 for criteria on Qualification of Hardware by Similarity. The final test verified Torque Margin shall be greater than zero (>0) based on the FS listed for the Acceptance / Qualification Test phase.

1.1 Program Phase	**1.2 Known Torque Factor of Safety (FS_k)**	**1.3 Variable Torque Factor of Safety (FS_v)**
Preliminary Design Review	2.00	4.0

Critical Design Review	1.50	3.0
Acceptance / Qualification Test	1.50	2.0

For those cases where high confidence does not exist in determination of worst case load or driving capability, a Safety Factor higher than that stated above may be appropriate. Factors of Safety shall be based on a confidence level determined from the quantity and fidelity of heritage and program test data. At the program PDR, a detailed plan to determine torque margin shall be presented. By CDR, it shall be demonstrated (see LEVR section 2.4.5.2) that the detail design complies with the program requirements as outlined in this section.

The required Factors of Safety shall be appropriately higher than given above if:

a. The designs involve an unusually large degree of uncertainty in the characterization of resistive torques.
b. The torque margin testing is not performed in the required environmental conditions or is not repeatable and has a large tolerance band.
c. The torque margin testing is performed only at the component level.

It is important to note that this torque margin requirement relates to the verification phase of the hardware in question. Conservative decisions shall be made during the design phase to ensure adequate margins will be realized. However, it is recognized that under some unique circumstances these specified Factors of Safety might be detrimental (excessive) to the design of a system. Any exceptions to these requirements shall be approved by the GSFC LDCM project office.

The minimum available driving torque for the mechanism shall be determined based on the FS listed above. The Torque Margin (TM) shall be greater than zero and shall be calculated using the following formula:

$$TM = \{T_{avail} / (FS_k \, \Sigma T_{known} + FS_v \, \Sigma T_{variable})\} - 1$$

Where:

Driving Torques:

T_{avail} = Minimum Available Torque or Force generated by the mechanism at worst case environmental conditions at any time in its life. If motors are used in the system, T_{avail} shall be determined at the output of the motor, not including gear heads or gear trains at its output based on minimum supplied motor voltage. T_{avail} similarly applies to other actuators such as springs, pyrotechnics, solenoids, heat actuated devices, etc.

Resistive Torques:

ΣT_{known} = Sum of the fixed torques or forces that are known and quantifiable such as accelerated inertias ($T=I\alpha$) and not influenced by friction, temperature, life, etc. A constant Safety Factor is applied to the calculated torque.

$\Sigma T_{variable}$ = Sum of the torques or forces that may vary over environmental conditions and life such as static or dynamic friction, alignment effects, latching forces, wire harness loads, damper drag, variations in lubricant effectiveness, including degradation or depletion of lubricant over life, etc.

2.4.5.4 Acceptance Requirements - For the acceptance testing of previously qualified hardware, the observatory and subsystem tests described in 2.4.5.2.b and 2.4.5.2.c shall be performed, except that the subsystem tests need be performed only at the nominal energy level. Adequate torque ratio (margin) shall be demonstrated for all flight mechanisms.

2.4.6 Pressure Profile Qualification

The need for a pressure profile test shall be assessed for all subsystems. A qualification test shall be required if analysis does not indicate a positive margin at loads equal to twice those induced by the maximum expected pressure differential during launch. If a test is required, the limit pressure profile is determined by the predicted pressure-time profile for the nominal trajectory of the particular mission.

Because pressure-induced loads vary with the square of the rate of change, the qualification pressure profile shall be determined by multiplying the predicted pressure rate of change by a factor of 1.12 (the square root of 1.25, the required qualification factor on load).

2.4.6.1 Demonstration - The hardware shall be qualified for the pressure profile environment by analysis and/or test. An analysis shall be performed to estimate the pressure differential induced by the nominal launch and reentry trajectories, as appropriate, across elements susceptible to such loading (e.g. thermal blankets, contamination enclosures, and housings of components). If analysis does not indicate a positive margin at loads equal to twice those induced by the maximum expected pressure differential, testing is required. Although testing at the subsystem level is usually appropriate, testing at the observatory level of assembly may be performed based on approval of the GSFC LDCM project office.

a. Test Profile - The flight pressure profile shall be determined by the analytically predicted pressure-time history inside the payload fairing for the nominal launch trajectory for the mission (including reentry if appropriate). Because pressure-induced loads vary as the square of the pressure rate, the pressure profile for qualification shall be determined by increasing the predicted flight rate by a factor of 1.12 (square root of 1.25, the required test factor for loads). The pressure profile shall be applied once.

b. Facility Considerations - Loads induced by the changing pressure environment are affected both by the pressure change rate and the venting area. Because the exact times of occurrence of the maximum pressure differential is not always coincident with the maximum rate of change, the pumping capacity of the facility shall be capable of matching the desired pressure profile within ± 5% at all times.

c. Test Setup - During the test, the subsystem shall be in the electrical and mechanical operational modes that are appropriate for the event being simulated.

d. Performance - Before and after the pressure profile test, the subsystem shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

2.4.6.2 Acceptance Requirements - Pressure profile test requirements do not apply for the acceptance testing of previously qualified hardware.

2.4.7 Mass Properties Verification

Hardware mass property requirements are mission-dependent and, therefore, are determined on a case-by-case basis. The mass properties program shall include an analytic assessment of the observatory's ability to comply with the mission requirements, supplemented as necessary by measurement.

2.4.7.1 Demonstration - The mass properties of the observatory shall be verified by analysis and/or measurement.

When mass properties are to be derived by analysis, it may be necessary to make some direct measurements of subsystems and components in order to attain the accuracy required for the mission and to ensure that analytical determination of observatory mass properties is feasible. Determination of the various subsystem properties shall be sufficiently accurate that, when combined analytically to derive the mass properties of the observatory, the uncertainties will be small enough to ensure compliance with observatory mass property requirements. If analytic determination of observatory mass properties is not feasible, then direct measurement shall be performed. The following mass properties shall be determined:

a. Weight, Center of Gravity, and Moment of Inertia - Weight, center of gravity, and moment of inertia are used in predicting observatory performance during launch, insertion into orbit, and orbital operations. The parameters are determined for all configurations to evaluate flight performance in accordance with mission requirements.

b. Balance - Hardware is balanced in accordance with mission requirements. Balance may be achieved analytically, if necessary, with the aid of direct measurements.

(1) Procedure for Direct Measurement - The usual procedure for direct measurement is to perform an initial balance before beginning the environmental verification program and a final balance after completing the program. One purpose of the initial balance is to ensure the feasibility of attaining the stipulated final balance. A residual unbalance of not more than four times the final balance requirement is the recommended objective of initial balance. Another reason for doing the initial balance prior to environmental exposures is to evaluate the method of attaching the balance weights and the effect of the weights on the operation of the hardware during the environmental exposures. Final balance shall be performed after completion of all environmental testing in order to properly adjust for all changes to weight distribution made during the verification program such as hardware replacement or redesign.

(2) Maintaining Balance - Changes to the hardware that may affect weight distribution shall be minimized after completion of final balance. The effects of such changes (including any disassembly, hardware substitution, etc.) on the residual unbalance of the hardware shall be assessed. That involves sufficient dimensional measurement and mass properties determination to permit a judgment as to whether the configuration changes have caused

the residual unbalance to exceed requirements. If so, additional balance operations shall be performed.

(3) Correcting Unbalance - To correct unbalance, weights may be attached, removed, or relocated. The amount of residual unbalance for all appropriate configurations shall be performed and recorded for comparison with the balance requirements of the verification specification. Balance operations include interface, fit, and alignment checks as necessary to ensure that alignment of geometric axes is comparable with requirements.

Balancing operations include measurement and tabulation of weights and mass center locations (referenced to hardware coordinates) of appendages, motors, and other elements that may not be assembled for balancing.

The data shall be analyzed to determine unbalance contributed by such elements to each appropriate configuration.

The facilities and procedures for balancing shall be fully defined at the time of initial balance, and sufficient exploratory balancing operations shall be performed to provide confidence that the final balance can be accomplished satisfactorily and expeditiously.

2.4.7.2 Acceptance Requirements - The mass property requirements cited above shall apply to all flight hardware.

SECTION 2.5

EMC

2.5 ELECTROMAGNETIC COMPATIBILITY (EMC) REQUIREMENTS

The general requirements for electromagnetic compatibility are as follows:

a. The observatory (spacecraft) and its elements shall not generate electromagnetic interference that could adversely affect its own subsystems and components, other observatories, or the safety and operation of the launch vehicle and launch site.

b. The observatory and its subsystems and components shall not be susceptible to emissions that could adversely affect their safety and performance. This applies whether the emissions are self-generated or emanate from other sources, or whether they are intentional or unintentional.

2.5.1 Requirements Summary

The EMC test requirements herein when performed as a set are intended to provide an adequate measure of hardware quality and workmanship. The tests shall be performed to fixed levels which are intended to envelope those that may be expected during a typical mission and allow for some degradation of the hardware during the mission. The levels shall be tailored to meet mission specific requirements, such as, the enveloping of launch vehicle and launch site environments, or the inclusion of very sensitive detectors or instruments in the observatory.

Thus tailored, the requirements envelope the environments usually encountered during integration and ground testing. However, because some observatories may have sensors and devices that are particularly sensitive to the low-level EMI ground environment, special work-around procedures may have to be developed to meet individual observatory needs.

2.5.1.1 The Range of Requirements - Table 2.5-1 is a matrix of EMC tests that apply to a wide range of hardware intended for launch either by an expendable launch vehicle (ELV). Tests shall be performed at the component, subsystem, and observatory levels of assembly. Not all tests apply to all levels of assembly or to all types of observatories. The Contractor shall select the requirements that fit the characteristics of the mission and hardware, e.g. a transmitter would require a different group of EMC tests than a receiver. Symbols in the hardware levels of assembly columns will assist in the selection of an appropriate EMC test program.

Once the EMC test program is selected, all flight hardware shall be tested. The EMC test program is meant to uncover workmanship defects and unit-to-unit variations in electromagnetic characteristics, as well as design flaws. The qualification and flight acceptance EMC programs are the same. Performance of both will provide a margin of hardware reliability.

Table 2.5-1
EMC Requirements per Level of Assembly

Type	Test Observatory* Observatory	Paragraph Number		ELV	Component	Subsystem/ Instrument	
CE	Dc power leads	2.5.2.1.a&c		X	RR	Sb	
CE	Power Leads	2.5.2.1.b		X	RR	-	
CE	Antenna terminals	2.5.2.1.f		X	R	-	-
RE	E-fields	2.5.2.2.c&d		X	RR	R	
RE	Observatory transmitters	2.5.2.2.e		X	-	-	**
RE	Spurious (transmitter antenna)	2.5.2.2.f		X	-		-
CS	Power line	2.5.3.1.a		X	RR	CS	
	Intermodulation products	2.5.3.1.b		X	R-	-	
CS	Signal rejection	2.5.3.1.c		X	R-	-	
CS	Cross modulation	2.5.3.1.d		X	R-	-	
CS	Power line transients	2.5.3.1.e		X	RR		
RS	E-field (general compatibility)	2.5.3.2.a		X	RR	R	

CE - Conducted Emission
CS - Conducted Susceptibility
R - Test to ensure reliable operation of observatory, and to help ensure compatibility with the launch vehicle and launch site
RE - Radiated Emission
RS - Radiated Susceptibility
* - Observatory, Mission, or highest level of assembly
** - Shall meet any unique requirements of launch vehicle and launch site for transmitters that are on during launch

The EMC tests are intended to verify that:

(1) The hardware will operate properly if subjected to conducted or radiated emissions from other sources that could occur during launch or in orbit (susceptibility tests).

(2) The hardware does not generate either conducted or radiated signals that could hinder the operation of other systems (emissions tests).

2.5.1.2 Testing at Lower Levels of Assembly - It is recommended that testing be performed at the component, subsystem, and observatory levels of assembly. Testing at lower levels of assembly has many advantages: it uncovers problems early in the program when they are less costly to correct and less disruptive to the program schedule; it uncovers problems that cannot be detected or traced at higher levels of assembly; it characterizes box-to-box EMI performance, providing a baseline that can be used to alert the project to potential problems at higher levels of assembly; and it aids in troubleshooting.

2.5.1.3 Basis of the Tests - A description of the individual EMC tests listed in Table 2.5-1, including their requirement limits and test procedures, are provided in paragraphs 2.5.2 through 2.5.4.7 Most of the tests are based on the requirements of MIL-STD-461C and 462, as amended by Notice 1, and MIL-STD-463A (1.7.8). Note: all references in this document to MIL-STD-462 assume reference to Notice 1.

For ELV launch, the EMI/EMC test levels may be modified based on the particular launch vehicle or launch site environments. Those requirements shall be established during coordination between the Contractor, GSFC LDCM project officeand the launch vehicle program office All modifications shall be approved by the GSFC LDCM project office.

2.5.l.4 Safety and Controls - During prelaunch and prerelease checkout, sensitive detectors and hardware may require special procedures to protect them from the damage of high-level radiated emissions. If such procedures are needed, they shall also be applied during EMC testing.

Except for bridgewires, live electroexplosive devices (EEDs) used to initiate such observatory functions as boom and antenna deployment shall be replaced by inert EEDs. When that is not possible, special safety precautions shall be taken to ensure the safety of the observatory and its operating personnel.

Spurious signals that lie above specified testing limits shall be eliminated. Spurious signals that are below specified limits shall be analyzed to determine if a subsequent change in frequency or amplitude is possible; if it is possible, the spurious signals shall be eliminated to protect observatory and instruments from the possibility of interference. Retest shall be performed to verify that intended solutions are effective

2.5.2 Emission Requirements

The following paragraphs on emission tests shall be used to implement the emission requirements of Table 2.5-1.

2.5.2.l Conducted Emission Limits - Conducted emission limits and requirements on power leads, as well as on antenna terminals, shall be applied to observatory hardware as defined below. The requirements do not apply to secondary power leads to subunits within the level of assembly under test unless they are specifically included in a hardware specification.

a. Narrowband conducted emissions on power, and power-return leads shall be limited to the levels specified in Figure 2.5-1.

Testing shall be in accordance with MIL-STD-461C and 462, test numbers CE01 and CE03, as applicable, with limits as shown in Figure 2.5-1.

b. A Conducted Emissions (CE) test to control Common Mode Noise (CMN) shall be required at the subsystem/component level. This frequency domain current test shall be performed on all non-passive components which receive or generate observatory primary power.

The purpose of the test is to limit CMN emissions that flow through the observatory structure and flight harness which result in the generation of undesirable electrical currents, and electro-magnetic fields at the integrated system level.

Specific CMN requirements shall be determined carefully from observatory hardware designs or mission scenario. Observatory which have analog or low level signal interfaces, low level detectors, and instruments that measure electromagnetic fields may be particularly sensitive to CMN. If mission requirements do not place stricter control on CMN, the limits of Figure 2.5-1a are suggested.

The CMN test procedure is the same as narrowband CE01/03 except that the current probe is placed around both the plus and return primary wires together.

c. Broadband conducted emissions on power, and power-return leads (shall be limited to the levels specified in Figure 2.5-2. Testing shall be in accordance with MIL-STD-461C and 462, test number CE03, with limits as shown in Figure 2.5-2.

d. (deleted)

e. (deleted)

f. Conducted emissions on the antenna terminals of observatory receivers, and transmitters in key-up modes shall not exceed 34 dB mV for narrowband emissions and 40 dB mV/MHz for broadband emissions.

Harmonics (greater than the third) and all other spurious emissions from transmitters in the key-down mode shall have peak powers 80 dB down from the power at the fundamental. Power at the second and third harmonics shall be suppressed by {50 + 10 Log(Peak Power in watts at the fundamental) dB}, or 80 dB whichever requires less suppression.

Testing shall be in accordance with MIL-STD-462, test number CE06. The test is conducted on receivers and transmitters before they are integrated with their antenna systems. Refer to MIL-STD-461C and MIL-STD-462 for additional details concerning this requirement.

2.5.2.2 Radiated Emission Limits - Radiated emission limits and requirements shall be applied to observatory hardware as defined in sections 2.5.2.2.a through 2.5.2.2.f below. Additional tests or test conditions shall be considered by the project if it appears that this may be necessary, for example, if the observatory receives at frequencies other than S-band (1.77 - 2.3 GHz).

a. (deleted

b. Radiated ac magnetic field levels produced by ELV-launched observatories and their subsystems shall be limited to 60 dB pT from 20 Hz to 50 kHz. This requirement may be deleted with project approval if subsystems or instruments are not inherently susceptible to ac magnetic fields.

If the free flyer observatories or their instruments contain sensitive magnetic field detectors or devices with high sensitivities to magnetic fields, more stringent limits on magnetic field emission may be required. Testing shall be in accordance with MIL-STD-462, test number RE04, with limits as defined above.

c. Unintentional radiated narrowband electric field levels produced by observatories shall not exceed the levels specified in Figure 2.5-9. Testing shall be in accordance with MIL-STD 461C and 462, test number RE02, with the test frequency range and limits revised as defined in Figure 2.5-9

d. Unintentional radiated broadband electric field levels produced by observatories shall not exceed the levels specified in Figure 2.5-10. Testing shall be in accordance with MIL-STD-461C and 462, test number RE02, with the test frequency range and the limits revised as defined in Figure 2.5-10.

e. Allowable levels of radiation from observatory transmitter antenna systems depend on the launch vehicle and launch site.

For an ELV launch, any unique requirements of the launch vehicle and launch site for transmitters that will be on during launch shall be met.

f. Radiated spurious and harmonic emissions from observatory transmitter antennas shall have peak powers 80 dB down from the power at the fundamental (for harmonics greater than the third). Power at the second and third harmonics shall be suppressed by {50 + 10 Log(Peak Power in watts at the fundamental) dB}, or 80 dB whichever requires less suppression. These are the same limits as those for conducted spurious and harmonic emissions on antenna terminals in paragraph 2.5.2.1.f. When the MIL-STD-462 test CE06 for conducted emissions on antenna terminals cannot be applied, test RE03 for radiated spurious and harmonic emissions shall be used as an alternative test. Refer to MIL-STD-461C and 462 for details.

2.5.2.3 Acceptance Requirements - The emission requirements of 2.5.2 shall also apply to all previously qualified hardware.

2.5.3 Susceptibility Requirements

The following paragraphs on susceptibility tests shall be used to implement the susceptibility requirements of Table 2.5-1.

2.5.3.1 Conducted Susceptibility Requirements - The following conducted susceptibility design and test requirements shall be applied to power leads and to antenna terminals of observatory hardware:

a. Conducted Susceptibility CS01-CS02 (Powerlines) - The tests shall be conducted over the frequency range of 30 Hz to 400 MHz in accordance with the limit requirements and test procedures of MIL-STD-461C and 462. If degraded performance is observed, the signal level shall be decreased to determine the threshold of interference. Above 50 KHz, modulation of the applied susceptibility signal is required if appropriate. If the appropriate modulation has not been established by component design or mission application, the following guidelines for selecting an appropriate modulation will apply:

(1) AM Receivers - Modulate 50 percent with 1000-Hz tone.

(2) FM Receivers - While monitoring signal-to-noise ratio, modulate with 1000-Hz signal using 10-kHz deviation. When testing for receiver quieting, use no modulation.

(3) SSB Receivers - Use no modulation.

(4) Components With Video Channels Other Than Receivers - Modulate 90 to 100 percent with pulse of duration 2/BW and repetition rate equal to BW/1000 where BW is the video bandwidth.

(5) Digital Components - Use pulse modulation with pulse duration and repetition rate equal to that used in the component under test.

(6) Nontuned Components - Use 1000-Hz tone for amplitude modulation of 50 percent.

b. Conducted Susceptibility CS03 (Two-Signal Intermodulation) - This test, which determines the presence of intermodulation products from two signals, shall be conducted on receivers operating in the frequency range of 30 Hz to 18 GHz where this test is appropriate for that type of receiver. The items shall perform in accordance with the limit requirements and the test procedures of MIL-STD-461C and 462 except that the operational frequency range of equipment subject to this test shall be increased to 18 GHz and the highest frequency used in the test procedure shall be increased to 40 GHz.

c. Conducted Susceptibility CS04 (Rejection of Undesired Signals) - Receivers operating in the frequency range from 30 Hz to 18 GHz shall be tested for rejection of spurious signals where this test is appropriate for that type of receiver. The items shall perform in accordance with the limit requirements and the test procedures of MIL-STD-461C and 462 except that the frequency range shall be increased to 40 GHz.

d. Conducted Susceptibility CS05 (Cross Modulation) - Receivers of amplitude-modulated RF signals operating in the frequency range of 30 Hz to 18 GHz shall be tested to determine the presence of products of cross modulation where this test is appropriate for that type of receiver. The items shall perform in accordance with the limit requirements and test procedures of MIL-STD-461C and 462 except that the operational frequency range of equipment subject to this test shall be increased to 18 GHz and the highest frequency used in the test procedure shall be increased to 40 GHz.

e. Conducted Susceptibility CS06 (Powerline Transient) - A transient signal shall be applied to powerlines in accordance with the procedures of MIL-STD-461C and 462. Because the applied transient signal shall equal the powerline voltage, the resulting total voltage is twice the powerline level. The transient shall be applied for a duration of 5 minutes at a repetition rate of 60 pps. The test shall be applied to the input power leads of all observatories.

2.5.3.2 Radiated Susceptibility Requirements - The following tests shall be applied to individual observatories and observatory subsystems. The tests are based on MIL-STD-461C and 462, as supplemented.

a. Radiated Susceptibility Test RS03 (E-field) - The observatory shall be exposed to external electromagnetic signals in accordance with the requirements and test methods of test RS03. Intentional E-field sensors on observatories that operate within the frequency range of the test shall be removed or disabled without otherwise disabling the observatory during the test. The test shall demonstrate that observatory (exclusive of E-field sensors) can meet their performance objectives while exposed to the specified levels. Modulation of the applied susceptibility signal is required. If the appropriate modulation has not been established by hardware design or mission scenario, then 50% amplitude modulation by a 100 Hz square wave shall be considered. When performing additional testing at discrete frequencies of known emitters, the modulation characteristics of the emitter shall be simulated as closely as possible.

(1) ELV-launched observatory:

- o 2 V/m over the frequency range of 14 kHz to 2 GHz.
- o 5 V/m over the frequency range of 2 to 12 GHz.
- o 10 V/m over the frequency range of 12 to 18 GHz; applicable only to observatory with a Ku band telemetry system.

The following levels shall be tested for informational purposes for observatory development and ground testing effects:

- o 10 V/m over the frequency range of 14 kHz to 2 GHz.
- o 20 V/m over the frequency range of 2 to 12 GHz.
- o 20 V/m over the frequency range of 12 to 18 GHz; applicable only to observatory with a Ku band telemetry system.

For ELV observatories, the EMI test levels (or frequency range) shall be increased if it is determined that onboard telemetry systems, another observatory, or other signals in space could expose a observatory to higher levels than the above test levels. Systems such as ground based radars are known to produce signals in space in excess of 2 V/m at frequencies at least as low as 400 MHz.

2.5.3.3 Acceptance Requirements - The susceptibility requirements of 2.5.3 shall apply to all previously qualified hardware.

2.5.4 Magnetic Properties*

An observatory whose magnetic properties or fields must be controlled to satisfy operational or scientific requirements, shall be tested at the component, subsystem, and observatory levels of assembly, as appropriate, and shall meet the following magnetic requirements (observatory with magnetic sensors, e.g., magnetometers, may have more stringent requirements):

2.5.4.l Initial Perm Test - The maximum dc dipole moment produced by a observatory and by each of its components following manufacture shall not exceed 3.0 and 0.2 AM^2 (dipole moment), respectively.

2.5.4.2 Perm Levels After Exposure to Magnetic Field - The maximum dipole moment produced by a observatory and each of its components after exposures to magnetic field test levels of 15 X 10^{-4} tesla shall not exceed 5.0 and 0.3 AM^2, respectively.

2.5.4.3 Perm Levels After Exposures to Deperm Test - The maximum dipole moment produced by a observatory and each of its components after exposures to magnetic field deperm levels of 30 X 10^{-4} tesla for observatory and 50 X 10^{-4} tesla for components shall not exceed 2.0 and 0.l AM^2, respectively.

2.5.4.4 Induced Magnetic Field Measurement - In order to obtain information for observatory magnetic design and testing, the induced magnetic field of components shall be measured while the components are turned off and exposed to a magnetic field test level of 0.6 X 10^{-4} tesla. The measurement shall be made by a test magnetometer that can null the magnetic test field.

2.5.4.5 Stray Magnetic Field Measurements - An observatory and each of its components shall not produce dipole moments due to internal current flows in excess of 0.5 and 0.05 AM^2, respectively.

2.5.4.6 Subsystem Requirements - Subsystems shall also be tested in accordance with the above requirements; however, the requirement limits shall be determined on a per case basis. The limits shall be designated between the levels for the observatory and those for components and shall depend upon the number of components in a subsystem and the number of subsystems in the observatory. Subsystem limits shall be designated such that the fully integrated observatory can meet its magnetic requirements.

2.5.4.7 Acceptance Requirements - The provisions for magnetic testing (2.5.4) shall apply to all previously qualified hardware.

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* Dc magnetics testing should be performed after vibration testing. This provides an opportunity to correct for any magnetization of the flight hardware caused by fields associated with the vibration test equipment.

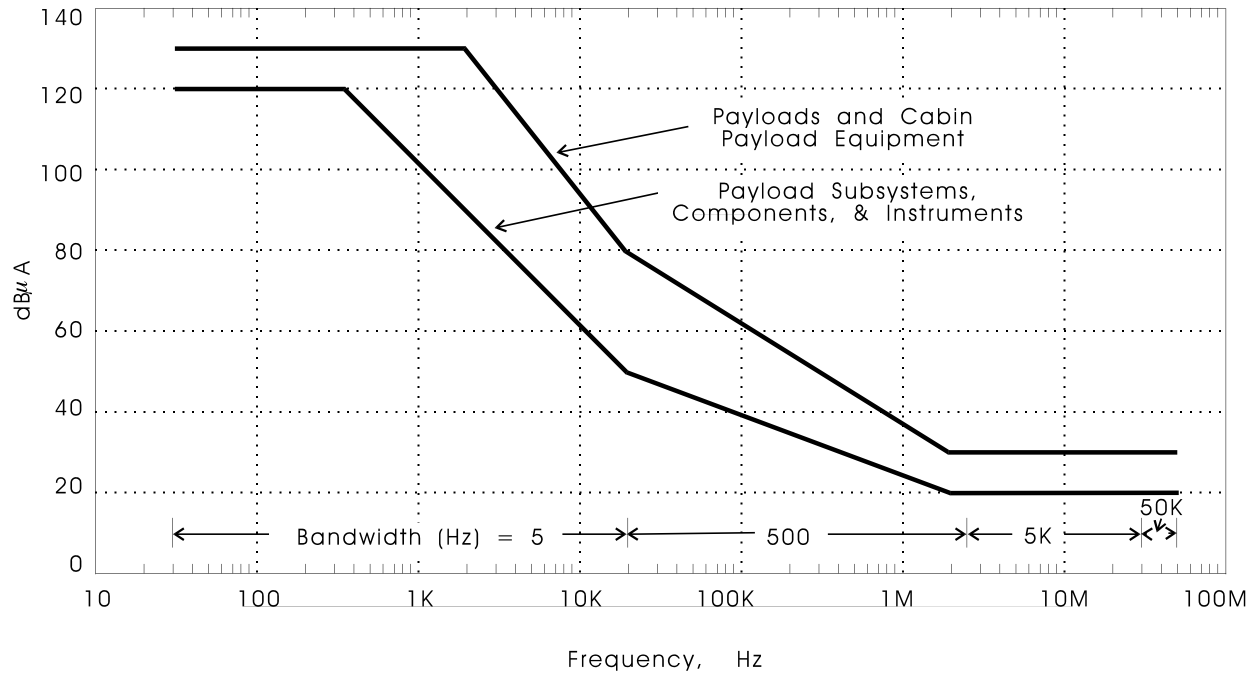

Figure 2.5-1 Narrowband Conducted Emission Limits on Observatory Power Lines

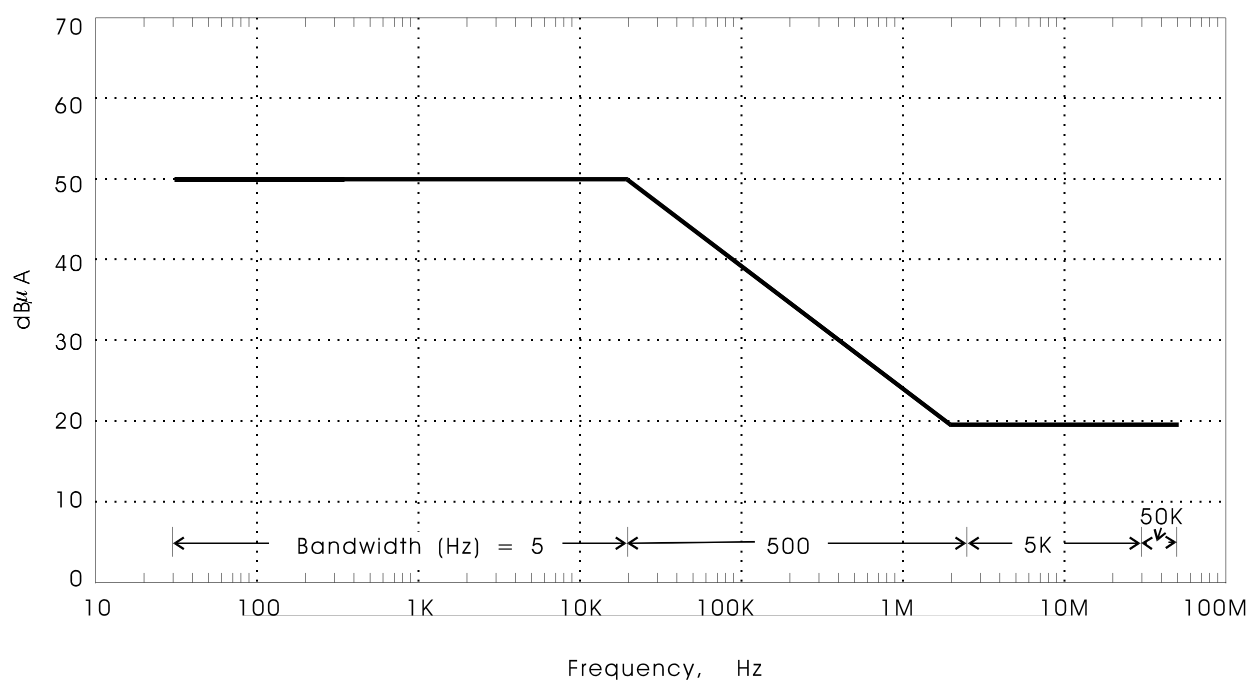

Figure 2.5-1a Common Mode Conducted Emission Limits on Primary Power Lines

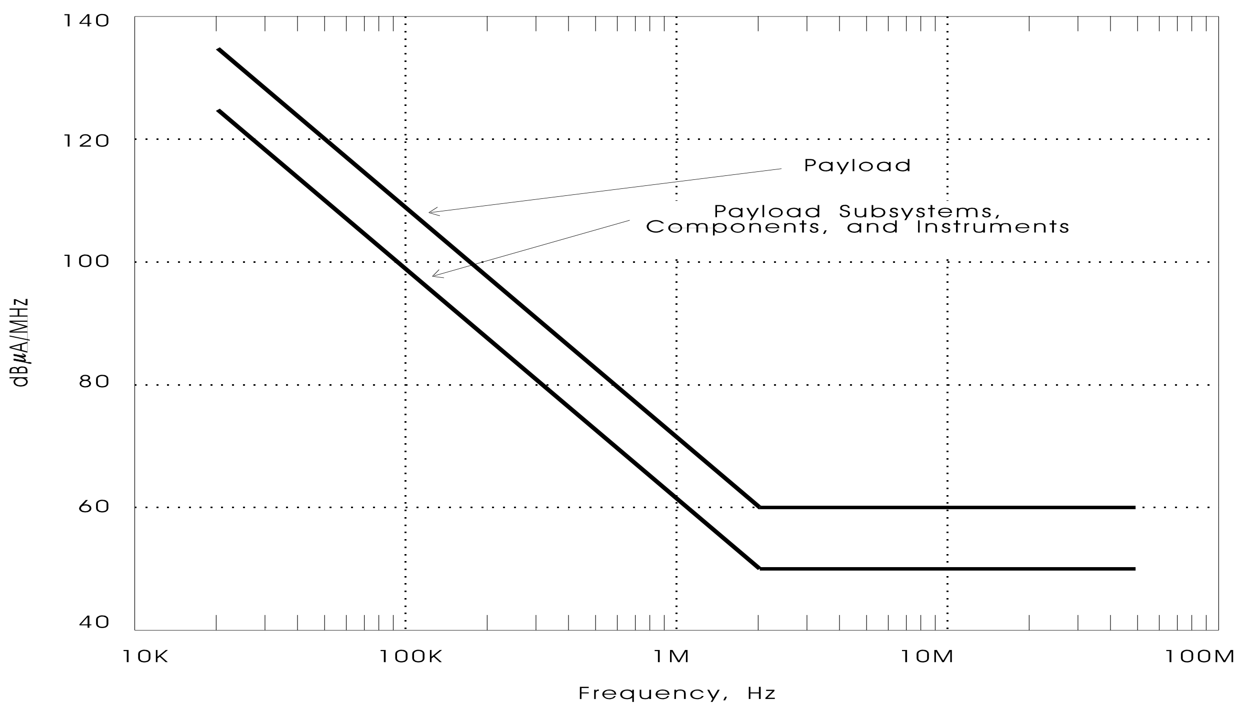

Figure 2.5-2 Broadband Conducted Emission Limits on Observatory Power Lines

SECTION 2.6

THERMAL

2.6 VACUUM and THERMAL REQUIREMENTS

The vacuum and thermal requirements herein apply to ELV observatories. An appropriate set of tests and analyses shall be selected to demonstrate the following observatory or observatory equipment capabilities.

a. The observatory shall perform satisfactorily within the vacuum and thermal mission limits (including launch .b. The thermal design and the thermal control system shall maintain the affected hardware within the established mission thermal limits during planned mission phases, including survival/safe-hold, if applicable.

d The quality of workmanship and materials of the hardware shall be sufficient to pass thermal cycle test screening in vacuum, or under ambient pressure if the hardware can be shown by analyses to be insensitive to vacuum effects relative to temperature levels and temperature gradients.

2.6.1 Summary of Requirements

Table 2.6-1 summarizes the tests and analyses that collectively will fulfill the general requirements of 2.6. Tests noted in the table may require supporting analyses. The order in which tests or analyses are conducted shall be determined by the Contractor and documented in the environmental verification plan, specification, and procedures. It is recommended, however, that mechanical testing occur before thermal testing at the systems level. Figure 2.6-1 shows the organization of the requirements and supporting information within this section of the LEVR.

The thermal cycle fatigue life test requirements of 2.4.2.1 also apply for hardware (e.g., solar arrays) susceptible to thermally induced mechanical fatigue.

The qualification and acceptance thermal-vacuum verification programs for passively controlled items are the same except that a 10°C temperature margin is added for qualification/protoflight testing and a 5°C margin is added for acceptance testing. For items controlled by active temperature control thermal systems, the margins are the same for qualification/protoflight and acceptance testing, as specified in 2.6.2.4.

Electronic card/piece part thermal analyses shall be performed to ensure that the GSFC Preferred Parts List (PPL) derated temperature limits and the allowable junction temperatures are not exceeded during qualification test conditions.

2.6.2 Thermal-Vacuum Qualification

The thermal-vacuum qualification program shall ensure that the observatory operates satisfactorily in a simulated space environment at more severe conditions than expected during the mission.

TABLE 2.6-1

VACUUM AND THERMAL REQUIREMENTS

Requirement	Observatory or Highest Practicable Level of Assembly	Subsystem including Instruments	Unit/ Component

Thermal-Vacuum[1,7]	T	T	2 T 2
Thermal Balance[1,3,7]	T and A	-	-
Leakage[6]	T	T	T

1. Applies to hardware carried in unpressurized spaces and to ELV-launched hardware.

2. Temperature cycling at ambient pressure may be substituted for thermal-vacuum temperature cycling if it can be shown by a comprehensive analysis to be acceptable. This analysis shall show that temperature levels and gradients are as severe in air as in a vacuum.

3 Consideration shall be given to environmental control of the enclosure.

6 Hardware that passes this test at a lower level of assembly need not be retested at a higher level unless there is reason to suspect its integrity.

7 Survival/Safehold testing is performed on that equipment which may experience (non-operating) temperature extremes more severe than when operating. The equipment tested is not expected to operate properly within specifications until the temperatures have returned to qualification temperatures.

T = Test required.

A = Analysis required; tests may be required to substantiate the analysis.

T/A = Test required if analysis indicates possible condensation.

T, A = Test is not required at this level of assembly if analysis verification is established for non-tested elements.

Note: Card level thermal analysis using qualification level boundary conditions is required to insure derated temperature limits, for example, junction temperature limits, are not exceeded.